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PRELIMINARY AIRWORTHINESS EVALUATION (PAE) OF THE OH - 58D HELICOPTER (AHIP)

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UNITED STATES ARMY AVIATION ENGINEERING FLIGHT ACTIVITY
EDWARDS AIR FORCE BASE, CALIFORNIA 93523

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The preliminary airworthiness evaluation of the OH-58D helicopter was conducted in Arlington, TX. ~~Texas, between 7 May and 9 August 1984.~~ The test required 56 flights for a total of 57 hours, of which 42.8 hours were productive. Handling qualities and vibration tests were conducted. The OH-58D in its present configuration is unsuitable for single pilot instrument meteorological conditions flight. Two enhancing characteristics were identified and concerned the capability to change pilot displays and switch preprogrammed radio frequencies ove

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without removing either hand from flight controls. Two deficiencies were identified: the warning displayed on the multifunction display masked the altitude display; and the copilot cyclic stick could be engaged while misaligned with the pilot's cyclic stick which would severely restrict cyclic travel. Additionally, 19 shortcomings and 5 instances of specification noncompliance were found. Keywords: Helicopters, Army aircraft, Observation aircraft.

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INTRODUCTION

BACKGROUND

1. The Army Helicopter Improvement Program (AHIP) has been designed to provide reconnaisance, security and target acquisition functions for an advanced attack helicopter fleet. The US Army contracted with Bell Helicopter Textron, Inc. (BHTI) to design, fabricate and qualify the OH-58D helicopter to satisfy this requirement. A mast mounted sight (MMS) was provided to improve scout survivability on the battlefield and to provide day/night operational capability. The US Army Aviation Systems Command (AVSCOM) tasked (ref 1, app A) the US Army Aviation Engineering Flight Activity (USAAEFA) to conduct a Preliminary Airworthiness Evaluation (PAE) of the OH-58D helicopter to provide data to substantiate the flight envelope for follow-on development and operational testing by government test agencies.

TEST OBJECTIVES

2. The objectives of this test were:

a. To determine contractual compliance with selected systems specification (ref 2, app A) requirements.

b. To provide limited assessment of the handling qualities and vibration characteristics.

DESCRIPTION

3. The test helicopter was a preproduction OH-58D (USA S/N 69-16285). The OH-58D helicopter is an extensively modified OH-58A helicopter. Three configuration changes were made during the course of the evaluation: 1) the horizontal stabilizer was extended 6 inches on each side, 2) roll Stability and Control Augmentation System (SCAS) gains were modified, 3) approximately 3 lb of weight was externally bonded to each main rotor blade. A detailed description of the OH-58D is contained in the draft operator's manual (ref 3, app A) and in appendix B. Major modifications to the OH-58D include:

a. a single T703-AD-700 (Allison model 250-C30R) turbine engine rated at 650 shaft horsepower, at sea level, standard day conditions (ref 4, app A);

b. a 4-bladed, foldable main rotor system;

c. an electronic stability and control augmentation system;

d. an extensively modified cockpit arrangement including multifunction displays and a multifunction keyboard for display control;

e. an electro-optical visionics system (MMS) mounted above the main rotor providing day/night infrared target acquisition and laser designation; and

f. a maximum gross weight of 4500 lb with a fuel capacity of 109.8 US gallons, a primary mission gross weight of 4045 lb and an alternate mission gross weight of 4150 lb.

TEST SCOPE

4. The flight tests for the PAE were flown at the BHTI flight test facility at Arlington, Texas between 7 May and 9 August 1984. The test required 56 flights for a total of 57 hours, of which 42.8 hours were productive. BHTI provided and maintained the aircraft and test instrumentation, and processed the test data. Testing was conducted in accordance with the test plan (ref 5, app A) and within the constraints of the draft operator's manual and the airworthiness release (ref 6). Handling qualities were evaluated with respect to the applicable requirements of MIL-H-8501A (ref 7) and the system specification. Test conditions are presented in table 1.

TEST METHODOLOGY

5. Flight test data were recorded on magnetic tape by an on-board BHTI instrumentation package (app C). Established flight test techniques were used (refs 8 and 9, app A). The test methods and data analysis are briefly discussed in appendix D. A Handling Qualities Rating Scale (HQRS) (fig. A, app D) was used to augment pilot comments relative to handling qualities. A Vibration Rating Scale (VRS) (fig. B, app D) was used to augment pilot comments relative to vibration. Pilot comments were recorded on cockpit data cards and a cockpit voice recorder.

Table 1. Test Conditions[1]

Type of Test	Gross Weight (lb)	Longitudinal Center of Gravity[2] (F.S.)	Density Altitude (ft)	Trim Calibrated Airspeed (kt)	Remarks
Control Positions in Trimmed Forward Flight	4300	109.9 (Mid)	6700	40 to 127	Level Flight IRP[3] Descent to V_{NE}[4]
Static Longitudinal Stability	4200	111.0	5840	44, 103	Level, Auto, and Climb
	3860	112.6	5850	45, 103	Level, Auto, and Climb (MMS[5] OFF)
	4210	110.0 (Mid)	6340	56, 108	Level and Climb (extended Stabilizer ON)
Static Lateral-Directional Stability	4240	111.1	6280	49, 102	Level Flight
	3800	112.5	6420	46, 102	Level, Auto, and climb (MMS OFF)
	4230	111.0	7060	107	Level Flight (Extended Stabilizer ON)
Maneuvering Stability	4210	111.2	5950	108	Steady Turns in Both Directions
Dynamic Stability	4390	110.8	6440	0, 102	Level, Climb Flight Mode - (SCAS[6] ON and OFF)
Controllability[7]	4450	111.0	1830	0	Hover
	4200[8]	110.6	2710		
	3940	112.3	1980		Hover (MMS OFF)
	4450	111.0	6400	102	Level Flight
	4210[8]	110.0 (Mid)	5880		
	3950	112.3	6540		Level Flight (MMS OFF)
	3900[8]	111.2	8350		
Slope Landing	4500	111.2	1642	0	10° left, right and nose-up 5° nose-down
Low Speed Flight	4490	110.5	560	0 to 35	10 ft Skid Height
Simulated Engine Failures	4390	111.10	5776	40, 75, 102	Level Flight and Climbs
Simulated SCAS Failures	4390	110.80	5620	70 to 115	Level Flight
			1800	0	Hover OGE[9]
Hydraulic Control Boost Failure	4380	111.30	6729	80 to 102	Level Flight
Instrument Meteorological Conditions (IMC)	4380	110.8	6400	60 to 115	Simulated IMC Tasks
Autorotational Landing Characteristics	4100	110.4	650-850	0 to 70	Original Unweighted Main Rotor Blades and Externally Weighted Main Rotor Blades

NOTES:
[1]Test conducted in the primary mission configuration (mast mounted sight installed, multipurpose lightweight missile system removed) at 395 RPM rotor speed with the stability and control augmentation system ON unless otherwise noted.
[2]Center of gravity locations were aft unless otherwise noted.
[3]IRP - Intermediate rated power
[4]V_{NE} - Never exceed airspeed
[5]MMS - Mast mounted sight
[6]SCAS - Stability and control augmentation system
[7]Longitudinal and lateral axis only in forward flight, longitudinal, lateral, and directional axis in hover.
[8]Extended horizontal stabilizer installed.
[9]OGE - Out of ground effect

RESULTS AND DISCUSSION

GENERAL

6. A limited handling qualities and vibration evaluation of the OH-58D helicopter was conducted at the Bell Helicopter Flight Test Facility, Arlington, Texas. During the course of the evaluation, three configuration changes were made to correct known deficiencies: 1) the horizontal stabilizer was extended on 24 May 1984 to correct dynamic stability problems with hydraulics OFF, 2) on 28 June 1984, changes in roll SCAS gains were implemented and reevaluated to correct a lateral pilot induced oscillation tendency, and 3) externally weighted main rotor blades were installed on 9 August 1984 to correct autorotational landing problems. The OH-58D in its present configuration is unsuitable for single pilot instrument meteorological conditions (IMC) flight. Two enhancing characteristics were identified and concerned the capability to change pilot displays and switch preprogrammed radio frequencies without removing either hand from the flight controls. Two deficiencies were identified. Warnings displayed on the multifunction display (MFD) masked the altitude display; and the copilot cyclic stick could be engaged while misaligned with the pilot's cyclic stick which would severely restrict cyclic travel. Additionally, 19 shortcomings were identified.

HANDLING QUALITIES

Control System Characteristics

7. The control system characteristics of the OH-58D were evaluated with rotors stopped and engine OFF. Hydraulic and electrical power were provided by a ground power unit. The evaluation was conducted with adjustable friction OFF and force trim ON. Control forces were measured with a hand held force gauge and recorded on data cards. A summary of control system characteristics is presented in table 2. The cyclic envelope is graphically depicted in figure 1, appendix E. Test results are presented in figures 2 through 4.

8. The cyclic had a total control throw of 10.3 inches, longitudinally, and 8.8 inches, laterally. Collective and pedal position had no effect on total cyclic movement.

9. Longitudinal and lateral control centering was positive but not absolute. Longitudinal cyclic had 0.2 inches of force trim system freeplay while the lateral cyclic had 0.1 inches of force trim system freeplay. There was no discernible control system

Table 2. OH-58D Control System Characteristics Summary[1]

Parameter	Control	Value	MIL-H-8501A Requirement	MIL-H-8501A Compliance	MIL-H-8501A Paragraph
Limit Control Force[2] (lb)	Longitudinal Lateral Directional Collective	15 lb 8 lb 32 lb 6.2 lb	8.0 lb 7.0 lb 15.0 lb 7.0 lb	Note 3	3.2.7 3.3.13 3.3.13 3.4.2
Trim System Freeplay (in.)	Longitudinal Lateral Directional	0.2 in. 0.1 in. 0.2 in.	— — —	— — —	— — —
Control Centering	Longitudinal Lateral Directional	Positive	Positive	Met	3.2.3
Breakout Force Including Friction (lb)	Longitudinal Lateral Directional Collective	Fwd 2.5 lb Aft 2.8 lb Left 1.8 lb Right 1.2 lb Left 14 lb Right 7.0 lb Up 4.4 lb Down 3.5 lb	0.5 to 1.5 lb 0.5 to 1.5 lb 3.0 to 7.0 lb 1.0 to 3.0 lb	Note 4	3.2.7 3.3.13 3.3.13 3.4.2
Force Gradient (lb/in.)	Longitudinal Lateral Directional	1.85 lb/in. 0.95 lb/in. 5.0 lb/in.	0.5 to 2.0 lb/in. 0.5 to 2.0 lb/in. Linear	Met Met Met	3.2.4 3.3.11 3.3.11
Control Travel (in.)	Longitudinal Lateral Directional Collective	10.3 in. 8.8 in. 6.8 in. 11.4 in.	— — — —	— — — —	— — — —

NOTES:

[1]Evaluated on the ground with engine and rotor static using ground power carts for electrical and hydraulic power.
[2]Measured with control displacement and trim set at full opposite directions.
[3]Inflight qualitative evaluation of the maneuvers specified in MIL-H-8501A, paragraphs 3.2.5 and 3.2.6 indicated that the limit control forces will meet specification.
[4]Inflight qualitative evaluation indicated that reduced breakout forces exist inflight and it is suspected that all axis will meet specification.

freeplay. Breakout plus friction forces were 2.5 lb forward, 2.8 lb aft, 1.8 lb left lateral, and 1.2 lb right lateral. Limit forces (cyclic displacement and trim set at maximum opposite directions) were 15.0 lb forward and aft, and 8 lb right and left.

10. Pedal centering was positive but not absolute with 0.2 inches of force trim system freeplay. Breakout forces were 14 lb left and 7 lb right. The average force gradient was 5.0 lb per inch of pedal displacement.

11. Collective control system characteristics were evaluated with cyclic and pedal controls at the neutral positions. Breakout forces were 4.4 lb, up, and 3.5 lb, down with adjustable friction OFF. With adjustable friction set to its maximum setting, breakout forces were 18.8 lb, up, and 17.5 lb, down.

12. Control system characteristics were qualitatively evaluated in flight. Breakout plus friction forces appeared reduced in flight and it is expected that all controls will meet breakout plus friction force specification. Cyclic control centering inflight appeared to be nearly absolute. When retrimming the cyclic control 1 inch, stick jump was minimal and not objectionable. The control system characteristics of the OH-58D helicopter are satisfactory.

Control Positions in Trimmed Forward Flight

13. The control positions of the OH-58D in trimmed flight were evaluated in level flight at the conditions shown in table 1. Test results conducted with the extended horizontal stabilizer are presented in figures 5 and 6, appendix E. Increased forward longitudinal control trim positions were required with increased forward airspeeds. Trim control position variations with airspeed showed no discontinuity, and adequate control margins were available. Pitch attitude varied from 1 degree nose up at 40 knots calibrated airspeed (KCAS) to 4 degrees nose down at maximum level flight airspeed. There was a large directional control trim shift (3 inches right) with collective movement between maximum climbing flight and autorotation at 50 KCAS (fig. 7). Additionally, any collective movement required a longitudinal cyclic trim change to maintain pitch attitude and a directional control trim change to maintain ball-centered flight. Control positions of the OH-58D in trimmed forward flight are satisfactory.

Static Longitudinal Stability

14. The static longitudinal stability characteristics of the OH-58D were evaluated in climbs, descents and level flight at the conditions shown in table 1, with and without the MMS and with both the small and extended stabilizor installed. Data are presented in figures 7 through 13, appendix E. The variation of longitudinal control position with airspeed was dependent on trim airspeed and power. In level flight the aircraft displayed weak to neutral static longitudinal stability. In climbing flight, static longitudinal stability was weak to neutral below trim airspeed and slightly positive above trim airspeed. This trend was reversed in autorotations. Longitudinal stability, as evidenced by longitudinal control position variation with airspeeds, was essentially the same for MMS ON or OFF and for extended stabilizer installed or removed even though trim control positions varied slightly. The static longitudinal stability characteristics of the OH-58D are essentially unchanged from the OH-58C (ref 10, app A) and are satisfactory.

Static Lateral-Directional Stability

15. Static lateral-directional stability characteristics of the OH-58D were evaluated in climbs, descents and level flight at the conditions shown in table 1, with MMS ON and OFF, doors ON and OFF and extended stabilizer installed and removed. Data are presented in figures 14 through 20, appendix E. At both airspeeds tested, the helicopter exhibited positive directional stability (increased left directional control for increase in right sideslip), and positive dihedral effect (increased right lateral control with increased right sideslip). The gradient of directional control position with sideslip angle in level flight was approximately 1 inch of pedal displacement per 30 degrees of sideslip at 49 KCAS, and was slightly steeper (1 inch of pedal displacement per 18 degrees of sideslip angle) at 107 KCAS. Sideforce cues were strong about trim as evidenced by the large change in roll attitude with sideslip at 107 KCAS. The effects of the MMS, doors or extended stabilizer was minimal on the lateral-directional stability characteristics. The static lateral-directional stability characteristics of the OH-58D are essentially unchanged from the OH-58C (ref 10, app A) and are satisfactory.

Maneuvering Stability

16. Manuevering stability was evaluated in left and right steady turns, pull ups and push overs at the conditions shown in table 1.

The data are presented in figures 21 and 22, appendix E. The stick-fixed maneuvering stability in steady turns, as indicated by the variation of longitudinal control position with normal acceleration (g), was positive until reaching a pitch rate of 7.5 degrees per second (°/sec). At 7.5°/sec or greater pitch rate, the right SCAS actuator was saturated resulting in pitch instability. Maintaining 100 knots indicated airspeed (KIAS) and 45 degrees bank angle in trim (±1/2 ball width), was very difficult using 1 to 2 inch longitudinal cyclic movements and 1/2 to 1 inch lateral cyclic movements resulting in airspeed variations of ±10 knots, bank angle variations of ±5 degrees and pitch attitude variations of ±5 degrees (HQRS 6). The pitch instability in steady turns of 45 degrees bank angle or greater and the resulting high pilot workload is a shortcoming.

Dynamic Stability

17. The short-term and long-term dynamic stability characteristics of the OH-58D were evaluated with the SCAS both ON and OFF at the conditions shown in table 1. Gusts were simulated in all control axes by single axis 1 inch control pulses and doublets, and by releases from steady heading sideslips. Tests were conducted with the extended stabilizer installed. The evaluation was conducted with the MMS both ON and OFF with no discernible difference in aircraft dynamic response. Dynamic stability was reevaluated with the new roll SCAS gain (ref para 24, app B) with no significant difference in aircraft dynamic response.

18. Representative time histories of the short-term longitudinal dynamic response with SCAS ON and OFF are shown in figures 23 and 24, appendix E. With SCAS OFF, the longitudinal short-term response was moderately damped while with SCAS ON it was highly damped. The longitudinal short-term response SCAS ON presented no problems in pitch attitude control. With SCAS OFF, pitch attitude was less predictable but still controllable. The longitudinal short-term response of the OH-58D is satisfactory.

19. Time histories of lateral-directional response are shown in figures 25 through 27. With SCAS ON at 50 KCAS, the lateral-directional response was moderately damped and resulted in primarily a roll oscillation with four overshoots before subsiding. With SCAS OFF at 100 KCAS, the lateral-directional response manifested itself as a 3-axis oscillation that was neutrally damped, however, at 50 KCAS in autorotation the lateral-directional response with SCAS OFF was divergent (time to double amplitude of 22 sec) with a period of 4.5 sec. The divergent tendency increased with airspeed. The airspeed at which the oscillation became difficult to control was approximately

80 KCAS. The SCAS ON lateral-directional dynamic response of the OH-58D is satisfactory. The maximum airspeed for SCAS OFF flight should be 80 KCAS.

20. The longitudinal long-term response characteristics with SCAS ON were evaluated by trimming the aircraft at the desired airspeed and then increasing or decreasing the airspeed using only the cyclic control. The cyclic control was then returned to the trim position and the aircraft response was recorded. Time histories of the longitudinal long-term responses are presented in figures 28 through 30, appendix E. The longitudinal long-term response at 102 KCAS was neutral to lightly damped with a period of approximately 26 sec. The long-term response was coupled to roll and yaw which had nearly the same periods as the pitch oscillation. The longitudinal long-term response at 102 KCAS was less damped than the OH-58C (ref 10, app A) but the long period of oscillation and relative difficulty in exciting the long-term response result in minimal pilot workload to maintain airspeed. At 50 KCAS in level or climbing flight with SCAS ON, the long-term response would self-excite and resulted in a divergent long-term response in both pitch and roll with a period of 24.5 sec. The response would diverge to a point requiring recovery within 3 cycles (approximately 59 sec). With SCAS OFF, the long-term response would diverge within one cycle to a point requiring recovery. The divergent long-term dynamic response of the OH-58D at 50 KCAS is a shortcoming.

Controllability

21. Longitudinal and lateral controllability tests were conducted in hover and in level flight at 102 KCAS in both the primary mission configuration (MMS installed) and in the low drag configuration (MMS removed) at the conditions shown in table 1. Directional control response was also evaluated in a hover in both configurations. Lateral controllability was reevaluated with the extended stabilizer installed and using the new roll SCAS gain changes (reference app B). Data are presented in figures 31 through 42, appendix E.

22. Directional controllability was evaluated in an out-of-ground effect (OGE) hover by making directional control step inputs of varying sizes in both directions and recording the aircraft response. A summary of directional control power (yaw attitude change after 1 sec), directional control response (maximum yaw rate) and directional control sensitivity (maximum yaw acceleration) is presented in table 3.

Table 3. Directional Controllability Summary

Test Conditions	Direction	Control Power (deg/in.)	Control Response (deg/sec/in.)	Control Sensitivity ($deg/sec^2/in.$)
3900 lb Gross Weight MMS OFF	Left	---	27	62
	Right	---	27	62
4210 lb Gross Weight MMS ON	Left	13	25	55
	Right	15	29	75
4500 lb Gross Weight MMS ON	Left	14	21	72
	Right	14.5	26	67

NOTE:

[1]Evaluated in out-of-ground effect hover only.

Yaw attitude predictability was good. Changing heading 90 degrees was easy requiring one pedal input to initiate yaw rate and two inputs to stop the turn at the desired heading (±5 degrees) (HQRS 3). Directional controllability of the OH-58D in a hover is satisfactory.

23. Longitudinal controllability was evaluated in an OGE hover and in level flight at 102 KCAS by making longitudinal control step inputs of varying sizes in both directions. A summary of longitudinal control power response and sensitivity is presented in table 4. Pitch attitude predictability was marginal. While accelerating to 40 KIAS and decelerating to a hover, pitch attitude control was difficult and required 1/2 to 1 inch longitudinal control inputs (1 per second) in order to effect and maintain a 10 degree (±3 degrees) pitch attitude change (HQRS 5). The unpredictable pitch response resulted in undesired altitude variations as well as pitch attitude fluctuations. The longitudinal control response resulting in unpredictable pitch attitude control is a shortcoming.

24. Lateral controllability was evaluated with the original roll SCAS gains and new roll SCAS gains as described in appendix B. A summary of lateral control power, response and sensitivity is presented in table 5.

Table 4. Longitudinal Controllability Summary

Configuration	Density Altitude (ft)	Airspeed (KCAS)	Control Power (deg/in.)	Control Response (deg/sec/in.)	Control Sensitivity (deg/sec^2/in.)
3940 lb Gross Weight MMS OFF	1980	0	6	8.3	70
3950 lb Gross Weight MMS OFF	6540	102	5	7.2	100
4210 lb Gross Weight MMS ON	1330	0	4.5	7.8	50
	6500	102	4	5 Fwd 7 Aft	40
4500 lb Gross Weight MMS ON	2330	0	6 Fwd 4.5 Aft	8.5	60 Fwd 40 Aft
4480 lb Gross Weight MMS ON	6320	102	4	7.5	40

Table 5. Lateral Controllability Summary[1]

Test Conditions	Control Power (deg/in.)	Control Response (deg/sec/in.)	Control Sensitivity (deg/sec^2/in.)
3900 lb Gross Weight MMS ON Hover	4	16	115
4200 lb Gross Weight MMS OFF 102 KCAS	5	18	135
4240 lb Gross Weight MMS ON 102 KCAS	4	19 Left 16 Right	125

NOTE:

[1]Extended horizontal stabilizer and new roll SCAS gains.

With the original roll SCAS gains, the lateral controllability was characterized by a high roll acceleration and peak roll rate followed by a rapid reduction in roll rate to about half the peak value. Roll attitude predictability was poor and during high workload tasks, where precise control of attitude was desired, a strong tendency to enter into pilot induced oscillations (PIO) existed. This prompted BHTI to change the roll SCAS gains. Initial acceleration and peak roll rate were not significantly different with the new roll SCAS gains. However, the character of the roll response was different as shown in figure 43, appendix E. The roll rate, after reaching peak value, did not reduce as rapidly and resulted in a period of nearly one second where roll rate remained nearly constant. This improved roll attitude predictability and eliminated the PIO tendency. Roll attitude predictability, though improved, was still marginal. When conducting lateral displacements with the new roll SCAS gains, roll attitude could be changed and maintained ($\pm$3 degrees) with 1/4 to 1/2 inch lateral cyclic inputs (2 per sec) (HQRS 5). Lateral controllability characteristics resulting in unpredictable roll attitude control is a shortcoming.

25. Controllability specification compliance is shown in figure 44, appendix E. Longitudinal and lateral control response did not meet the requirements of the system specification, paragraph 3.2.1.4.2 in that the plot of control sensitivity versus rate damping did not fall within the plot area required.

Slope Landing Characteristics

26. Slope landing characteristics were evaluated in the primary mission configuration at the conditions shown in table 1. Winds were from upslope at 15 knots. Slope landings were conducted from a hover up to 10 degrees nose up, left skid-up and right skid-up and up to 5 degrees nose down. Representative time histories of left and right skid up slope landing and takeoff are presented in figures 45 and 46, appendix E. The minimum control margin (0.4 in. right lateral cyclic remaining) existed during right skid upslope landings, whereas the highest pilot workload was during left skid upslope landings. Conducting a 10 degree right skid upslope landing required 3 inches of right lateral cyclic into the slope to maintain roll control during touchdown. Longitudinal cyclic was moved 1 inch aft while directional controls were moved approximately 0.5 inch right. This resulted in a smooth controlled touchdown (HQRS 3). The left skid upslope landing required approximately 4 inches of left lateral cyclic with longitudinal cyclic and directional control movements nearly the same as with the right skid upslope landing. Control movements were more oscillatory in nature and resulted in imprecise roll attitude control (HQRS 5). The higher pilot workload with the left skid upslope is attributable to the high pilot workload at critical relative wind azimuth (ref para 27). The nose up slope landing was the easiest. Due to the nose high hovering attitude, little actual attitude change and corresponding control movements were necessary to land the aircraft (HQRS 2). Nose-down slope landings were easy, but were limited to 5 degrees due to stinger ground contact. Again this is attributable to the nose high hovering attitude. In general, slope landings in the OH-58D are considered easier to perform with more positive aircraft control throughout the maneuver than in the OH-58C and are satisfactory. Slope landing characteristics did not meet the requirements of the System Specification (para 3.2.1.4.5) in that nose-down slope landings are limited to 5 degrees which is 5 degrees less than the requirement.

Low-Speed Flight Characteristics

27. The low-speed flight tests of the OH-58D were conducted to determine control margins and handling characteristics in low-speed flight by simulating winds from various relative azimuths. Tests were conducted at a skid height of 10 feet, using a ground pace vehicle, in surface winds of less than 5 knots at the conditions shown in table 1. Test results are presented in figures 47 and 48, appendix E.

28. At all azimuths tested with SCAS ON, control margins were adequate with the minimum control margins occurring at an azimuth of 90 deg at 30 knots true airspeed (KTAS) where 22% left directional control margin was remaining. In left sideward flight, between 15 to 25 KTAS, pedal activity was high and required 2 to 3 inch movements once per second to maintain heading ±5 degrees (HQRS 5). Pilot workload was minimal at all other relative azimuths tested and resulted in heading remaining within ±3 degrees with only 1 inch pedal movements every 1 or 2 seconds (HQRS 3). The low-speed flight characteristics of the OH-58D with SCAS ON are significantly improved over the OH-58C, however, the high pilot workload encountered in left sideward flight between 15 and 25 KTAS is a shortcoming. Low-speed flight characteristics did not meet the requirements of the system specification (para 3.2.1.4.1) in that left sideward flight required unusual control manipulations and pilot workload was not minimal.

Directional Trimmability

29. Directional trimmability was evaluated concurrently with trim control position, static stability, maneuvering stability and instrument flight capability tests. The general trends of directional trim changes required in forward flight are shown in figures 1 and 2. Small changes in power, airspeed, or turn rate required significant changes in directional position. Due to the high side forces associated with sideslips (ref para 15) and the directional trim change with airspeed, a 5 knot airspeed change in forward flight would result in a full ball width out-of-trim condition until the pedals were retrimmed. The side forces would fluctuate from side to side for approximately 2 to 3 cycles during the retrimming process which required pedal inputs of 1/4 to 1/2 inch (1 per 3 sec) to maintain directional trim ($\pm1/2$ ball width) (HQRS 5). The fluctuating high side forces encountered during this retrimming sequence were vertigo inducing and led to increased workload in IMC flight. The difficulty in maintaining directional trim when making small changes in flight conditions is a shortcoming.

Tail Rotor Effectiveness

30. A limited evaluation of tail rotor effectiveness was conducted during three maneuvers: 1) masking and unmasking; 2) 40 KIAS approach into the wind with a right turn to a left crosswind hover termination; and 3) 40 KIAS approach into the wind with a 180 degree right turn to a downwind hover. Tests were conducted at 4200 lb gross weight, aft center of gravity (cg), 680 feet

FIGURE 1. DIRECTIONAL TRIM CHANGES IN TURNS

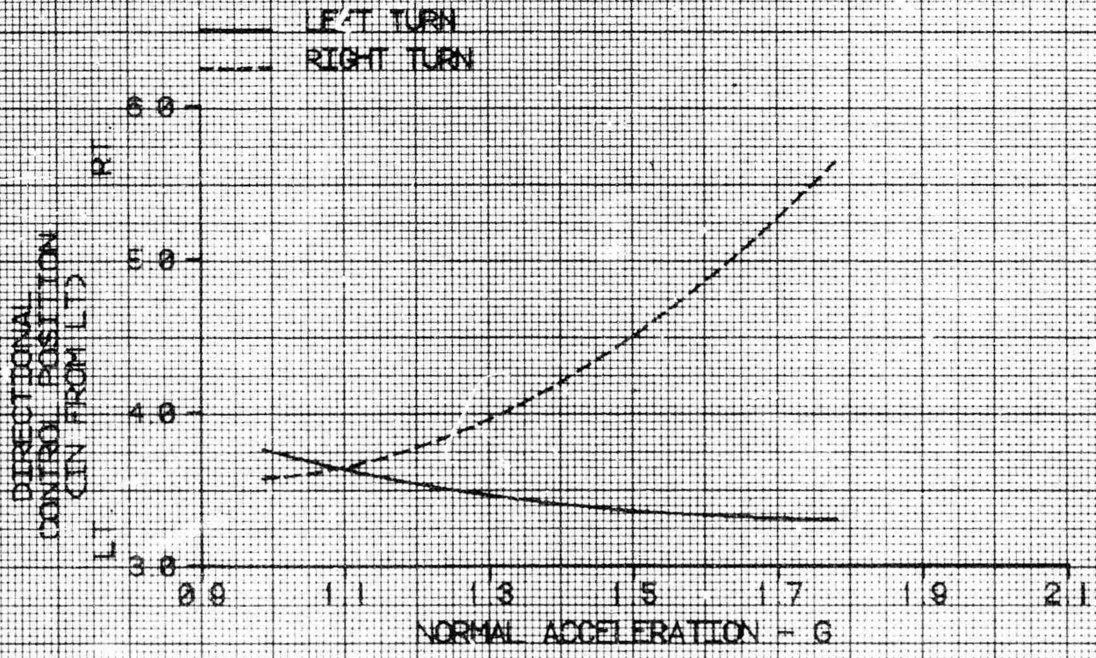
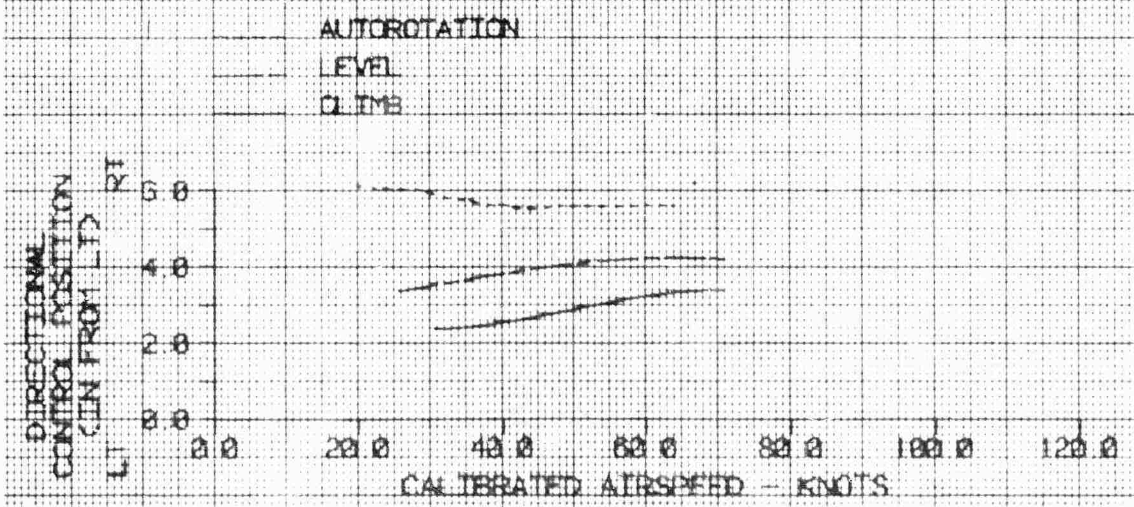


FIGURE 2. DIRECTIONAL TRIM CHANGES WITH POWER AND AIRSPEED



density altitude and 15 knot winds. Main rotor speed was maintained at 395 rpm. The most critical maneuvers were the approaches to a crosswind hover and to a downwind hover. Representative time histories of the most critical maneuvers are shown in figures 49, and 50, appendix E. Yaw rate fluctuations were experienced but could be quickly arrested with 1/4 to 1/2 inch pedal inputs. No loss of tail rotor effectiveness was experienced and yaw control was responsive and predictable. Tail rotor effectiveness as determined by the maneuvers tested is satisfactory.

Instrument Flight Capability

31. Single pilot instrument flight capability was evaluated in light turbulence at the conditions shown in table 1. Two pilots were on board for safety, however, for evaluation purposes, the copilot was restricted from accomplishing any IMC tasks. The pilot was flying with a hood to simulate IMC. Maneuvers conducted during the evaluation included level flight, level turns, climbs and descents, climbing and descending turns, a ground controlled approach and a tactical instrument approach using a doppler waypoint. Characteristics that significantly impacted on instrument flight capability could be divided into two categories: 1) aircraft flying qualities and 2) systems operation and displays. Aircraft flying qualities that affected instrument flight capability were the divergent long-term dynamic stability characteristics (ref para 19) and the difficulty in maintaining directional trim (ref para 29). When conducting a tactical instrument approach at 60 KIAS, the divergent long-term response required constant pilot attention to maintain aircraft control. Longitudinal control inputs of 1/4 to 1/2 inch were required (1 per 2 sec) to maintain airspeed ($\pm$5 knots) and 1/4 inch directional inputs were required (1 per 3 sec) to maintain aircraft in trim ($\pm$1/2 ball) in level flight (HQRS 6). The difficulty in maintaining directional trim and the resultant high side forces were vertigo inducing in IMC flight. Systems operation and display characteristics that affected IMC flight were the transponder location (ref para 50), the readability of transponder codes (ref para 51), the vertical speed indicator (ref para 53), and the standby instruments location (ref para 54). Additionally, the displayed altitude, being a digital display, provides no trend information and in conjunction with the erratic vertical speed indicator, made altitude control very difficult requiring 1/2 inch collective movement (1 per 10 sec) to maintain altitude ($\pm$100 feet) (HQRS 5). The multifunction display (MFD) can display a screen of information concerning communication and navigation or a screen of primary flight instruments. The necessity to update or change navigational information and to change radio frequencies requires the pilot to remove the primary flight instrument display from his MFD.

The difficulty of transitioning and using the standby flight instruments required the pilot to essentially fly "by the seat of the pants" while the pilot has the navigation or communication pages selected on the MFD. The divergent long-term response, imprecise altitude control, and the vertigo inducing sideforces associated with the difficulty in maintaining directional trim all contribute to a high pilot workload in IMC flight. The difficulty in performing other IMC cockpit tasks (transponder operation, navigation and communication) add to the pilot workload in IMC flight. The high pilot workload associated with aircraft flying qualities and pilot operation of systems and displays make the OH-58D unsuitable for single pilot IMC flight required by paragraph 3.1.7.1 of the system specification.

System Failures

Simulated Engine Failure:

32. Simulated sudden engine failures were evaluated with the standard (unweighted) main rotor blades at the conditions shown in table 1. Sudden engine failures were evaluated at 50, 75, and 100 KCAS at incremental engine powers varying from that required for a 500 fpm descent to intermediate rated power (IRP). Sudden engine failures were simulated by a rapid reduction in the throttle to flight idle with controls fixed until recovery was necessary. A time history of the most critical conditions (100 KCAS and IRP) is presented in figure 51, appendix E. The predominant characteristics of all simulated engine failures tested was minimal pitch and roll attitude changes coupled with a large yaw acceleration and consequent yaw attitude change prior to collective reduction. The large yaw acceleration caused immediate inertially induced lateral cyclic input even though the pilot was trying to maintain controls fixed. However, the cyclic movement was in a direction to minimize roll attitude changes. At 100 KCAS, IRP, rotor speed decayed at a rate of 55 rpm per second. Minimum rotor speed attained was 317 rpm with a collective reaction delay time of 0.9 seconds. Stabilizing in a steady state autorotation was easy and required 3 inches of aft longitudinal cyclic, 2 inches of right lateral cyclic and 2.5 inches of right pedal (HQRS 3). Rotor speed increase after collective reduction was slower than the decay with no apparent tendency to overspeed. The large yaw acceleration upon throttle reduction was the first indication of engine failure and was followed by low rotor speed visual and audio warning. The short collective delay time allowable, following a sudden engine failure, in order to remain above the minimum rotor speed of 310 rpm is a shortcoming. Collective delay time allowable following a sudden engine failure did not meet the requirements of the System Specification, paragraph 3.2.1.4.b in that collective delay time

of 0.9 seconds at 100 KCAS, IRP, was less than the required 1.2 seconds.

Stability and Control Augmentation System Failures:

33. SCAS failures were evaluated in an OGE hover and in forward flight to never exceed airspeed (V_{NE}) at the conditions shown in table 1. SCAS disengagements, single-axis hardovers and SCAS OFF decelerations from V_{NE} were evaluated. SCAS disengagements were simulated by depressing the cyclic SCAS disengagement button. Single axis hardovers were simulated through computer control. Special software accessed through the keyboard would externally send a hardover signal to the SCAS actuator upon command. The actuator and direction of hardover could be selected from the cockpit keyboard. A time history of SCAS disengagement at 110 KIAS (5 knots below V_{NE}) is presented in figure 52, appendix E. A time history of the most critical SCAS hardover (right actuator forward) at 110 KIAS is presented in figure 53. Both SCAS disengagement and single axis hardovers were benign resulting in minimal attitude changes. The maximum aircraft attitude change within the first two seconds after failure was during the pitch SCAS hardover which resulted in 6 degrees nose down, and 5 degrees right roll. Aside from the caution/warning system audio and visual indications of SCAS failure, the most predominant cockpit cue of a SCAS failure was the change in flying qualities (ref para 19) which were still controllable. The aircraft response to stability and control augmentation system failures in the OH-58D are satisfactory.

Hydraulic System Failures:

34. Hydraulic system failures were evaluated with the small stabilizer and extended stabilizer installed at the conditions shown in table 1. Hydraulic system failure was simulated by switching off the main hydraulic system with the test article backup hydraulic system disabled. Hydraulics were disabled at airspeeds up to V_{NE} then decelerations were performed to below 80 KIAS (the BHTI recommended maximum airspeed for continued hydraulics off flight). Additionally, hydraulics off landings were conducted. Time histories of decelerations from V_{NE} to 80 KIAS are shown in figures 54 and 55, appendix E.

35. With the small stabilizer installed, initial aircraft reaction to failure of the hydraulics system was negligible. However, the easily excitable divergent 3-axis oscillation (ref para 19) required control inputs in all axis to maintain aircraft control. Maintaining desired pitch attitude ($\pm$5 degrees) roll attitude

(±5 degrees) and in trim ($\pm1/2$ ball) required one inch longitudinal cyclic movements, 1 per second, 0.5 inch lateral cyclic movement, 1 every 2 seconds, and 0.5 inch pedal movement, 1 every 2 seconds (HQRS 7). Large cyclic forces were required (up to 19 lb) predominantly in the forward direction. Longitudinal cyclic forces were very imbalanced and became fatiguing during continued hydraulics off flight.

36. With the extended stabilizer installed, control movement and control forces were decreased during the deceleration. Longitudinal control forces were still imbalanced with a predominant forward cyclic force required throughout the forward flight regime and continued hydraulics off flight was still fatiguing. However, as airspeed decreased for landing hydraulics OFF, the forces decreased such that conducting a run-on landing or termination to a hover (and subsequent landing from a hover) was easily accomplished and required less than 0.5 inch cyclic movements and 1 inch pedal movement (HQRS 5). The imbalanced longitudinal cyclic forces in forward flight with hydraulics off causing pilot fatigue is a shortcoming.

Autorotational Landing Characteristics

37. Autorotational landing characteristics were evaluated with the original unweighted main rotor blades and the externally weighted main rotor blades installed. Both evaluations were conducted with a BHTI pilot as the safety pilot in the copilot's seat. Test conditions are shown in table 1. Both hovering and straight-in autorotations were evaluated.

38. Initial evaluation of the autorotative landing characteristics was accomplished during pilot training with the unweighted main rotor blades installed. A representative time history of a hovering autorotation is presented in figure 56, appendix E. Hovering autorotations were initiated from a 3 to 4 foot hover height with head winds of 5 knots. At initial throttle reduction, approximately 1 inch of left lateral cyclic, 0.2 inches of aft longitudinal cyclic, and 1.5 inches of right pedal were required to maintain heading within ±5 degrees and position over the ground within ±1 foot (HQRS 5). Rotor speed decay was rapid and resulted in little margin for error in collective application timing and rate in order to touchdown before the usable rotor energy was expended. Hovering autorotations in the OH-58D are significantly different than in the OH-58C in that apparent rate of descent would build faster, rotor energy appeared to decay faster, and initial control movement to maintain position over the ground was greater.

39. Straight-in autorotations with the unweighted main rotor blades installed were initiated from an altitude of 700 feet above ground level (AGL) at approximately 65 to 70 KIAS. Autorotational entry was accomplished by lowering collective to full down, rolling the throttle to flight idle position, and reducing airspeed to 60 to 65 KIAS. A representative time history is presented in figure 57, appendix E. Initial descent rate was high (2100 fpm) and rotor rpm decayed slightly (382 rpm) during the descent. Flare altitude was critical in that too high an altitude would result in loss of flare affectiveness at an altitude too high to effectively cushion the landing and too low of a flare altitude would result in either very long ground runs (greater than 100 ft) or tail stinger contact. Initiation of the flare occurred at approximately 50 to 60 feet AGL with maximum pitch attitude occurring at approximately 35 feet AGL. Flare effectiveness was minimal and resulted in little perceived decrease in forward velocity and descent rate with the apparent approach angle remaining the same. Forward cyclic and initial collective application were nearly simultaneous. Collective application was a two step movement. Initial collective application of approximately 2 inches was held 0.5 seconds and was immediately followed by sufficient collective application (approximately 5 inches) to cushion the touchdown. Autorotational landings with descent rate below 200 fpm at touchdown and ground run below 50 feet could not be accomplished. Precise judgement of flare initiation (±10 ft of optimum), flare attitude (23 degrees nose up ±3 degrees), and application of cyclic and collective to cushion the touchdown was required. Even properly executed autorotational landings resulted in high touchdown speeds and long ground runs (HQRS 7).

40. Hovering and straight-in autorotational landings were reevaluated with the externally weighted main rotor blades installed. A BHTI pilot acted as safety pilot during the evaluation. Hovering autorotational landing characteristics were significantly improved from the initial evaluation. Although control movements at the throttle reduction were essentially the same, rotor speed did not decay as rapidly and collective application timing and rate was not as critical in order to effect a smooth touchdown.

41. Straight-in autorotations with the externally weighted main rotor blades were initiated at the same flight conditions as the initial evaluation. A representative time history is presented in figure 58, appendix E. The externally weighted main rotor blades provided a significant improvement in the autorotational landing characterisitcs of the OH-58D. Flare altitudes from 50 to 100 feet were attempted and resulted in successful landings.

The flare attitude required was slightly reduced (approximately 20 degrees nose up) and resulted in increased flare effectiveness (i.e., rate of descent was checked in the flare). Initial collective application was less critical and could be accomplished while still in the flare attitude to assist in decelerating the aircraft. Even after initial collective application in the flare, sufficient rotor energy remained to effect a smooth and controlled touchdown with ground speeds of approximately 18 knots. Autorotational landings with the externally weighted main rotor blades installed could consistently and safely be accomplished with an adequate window of acceptable entry conditions (flare altitude of 50 to 100 feet and flare attitude of 15 to 25 degrees nose up) and acceptable landing techniques (HQRS 5). The autorotational landing characteristics of the OH-58D with externally weighted main rotor blades installed are satisfactory.

VIBRATION

42. Vibration characteristics at the aircraft center of gravity, pilot seat, and copilot seat were evaluated in conjunction with level flight control position in trimmed forward flight and low speed flight characteristics tests at the conditions shown in table 1. Data are presented in figures 59 through 61, appendix E. All vibrations throughout the flight envelope were minimal (VRS 1). Of particular note was the lack of vibrations when accelerating or decelerating through the effective translational lift regime (15 to 20 knots). However, the 4 per revolution (26.33 Hz) vibration at the copilot seat did not meet the requirements of MIL-H-8501A, paragraph 3.7.1.(b) in that the vibration level from 100 to 110 knots exceeded 0.15g. The vibration levels are acceptable.

COCKPIT EVALUATION

General

43. The cockpit of the test OH-58D helicopter was qualitatively evaluated throughout the test program.

Changing Displays

44. Displays on the pilot's MFD could be changed using the DISPLAY SELECT SWITCH on the pilot cyclic grip as shown in figure 16, appendix B. All displays, with the exception of the INITIAL PAGE, could be called up by the pilot without removing his hands

from the controls. The ease of changing displays on the pilot's MFD without removing either hand from the controls is an enhancing characteristic.

Changing a Radio Frequency

45. Thirty frequencies can be preprogrammed and five additional frequencies can be input manually. Changing to a different radio or a different frequency was easy and efficient as long as the frequency was preprogrammed. Without removing his hands from the controls, the pilot could change radios by using the RADIO SELECT SWITCH and change frequencies up and down by using the COMMUNICATION CONTROL SWITCH located on the pilot collective control head as shown in figure 17, appendix B. Changing frequencies to a frequency that is not preprogrammed is a little more difficult in that the multifunction keyboard (MFK) has to be used. The ease of changing to a different radio or to a frequency that is already programmed without taking hands from the collective or cyclic is an enhancing characteristic.

Multifunction Display Warning

46. The MFD displays altitude in the upper right hand corner as shown in photo 1, appendix F. Warning indications are displayed in the same location as shown in photo 2, appendix F. Warning indications have priority over altitude displays. Each warning requires some immediate action which may or may not allow the pilot the opportunity to acknowledge the warning to clear the screen. At night when visual determination of height above ground is extremely difficult, the altitude is critical for a safe landing. In this type of emergency, transitioning to the standby altimeter is unacceptable. The location of the warning display on the MFD which blocks out the altitude display is a deficiency.

Copilot/Observer Cyclic Stick

47. The copilot/observer (CPO) cyclic stick has the capability to be locked in position to prevent the CPO from inadvertently moving it. It is possible to engage the CPO cyclic stick in a different location from the pilot's cyclic as shown in photo 3, appendix F, which results in restricting movement of both sticks due to their misalignment. The possibility of restricted control travel as a result of cyclic stick misalignment is a deficiency. The following WARNING should be placed in the operator's manual:

WARNING

Engaging or disengaging the CPO cyclic stick is prohibited in flight, except during emergency conditions, due to possible misalignment of the pilot/CPO cyclic controls which will severely restrict cyclic movement. During an emergency which requires the CPO cyclic stick to be engaged, the pilot's cyclic should be returned as close as possible to the neutral position before engaging the CPO cyclic stick.

Engine and Transmission Vertical Scale Display

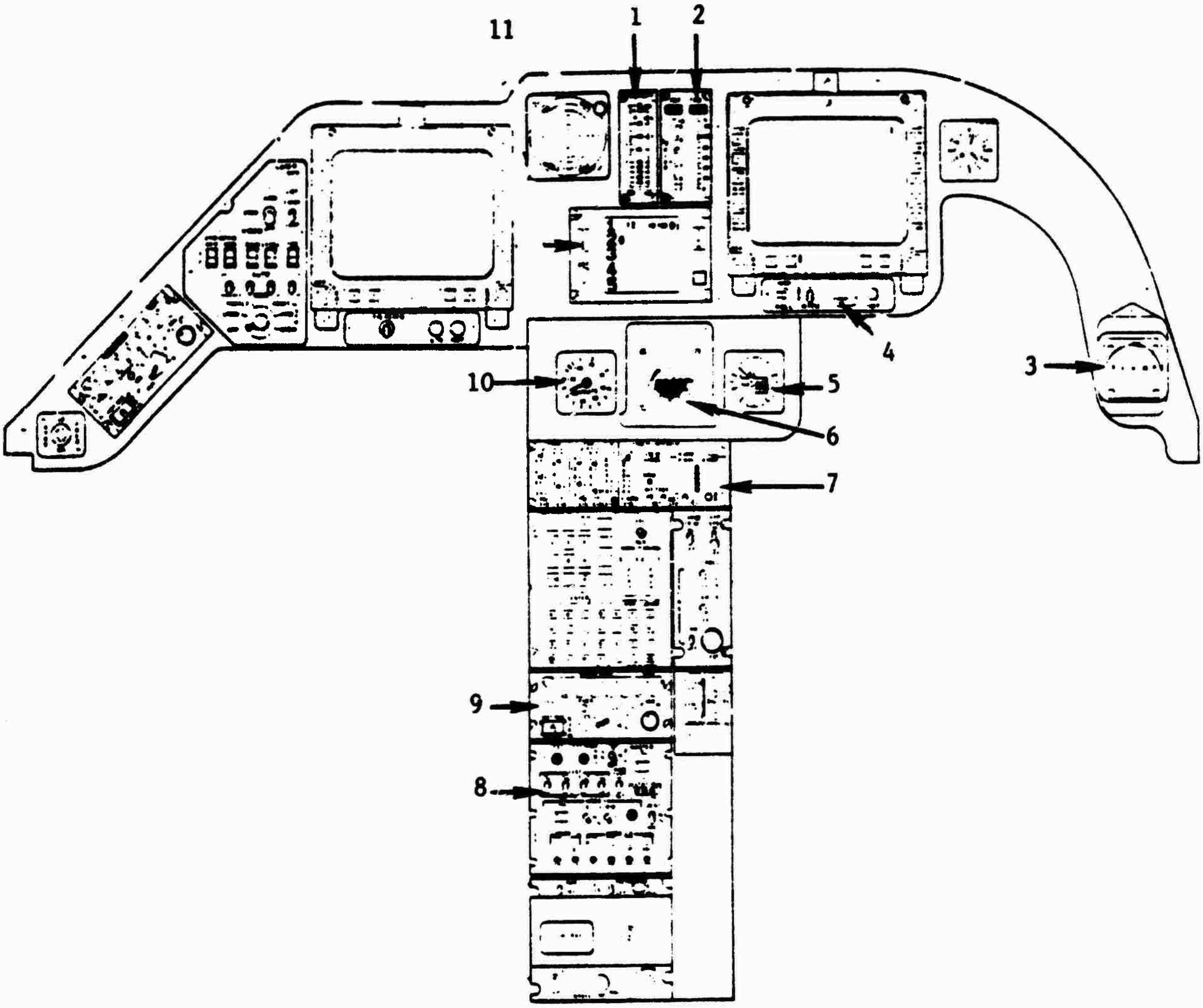
48. The limits on engine and transmission temperature and pressure guages are depicted on the side of the vertical strip as shown in photo 24, appendix B. At a glance, the engine and transmission temperature and pressure indications are very difficult to read and interpret. The difficulty in reading the gauges requires the pilot to concentrate on the gauges momentarily to get an accurate assessment of the engine and transmission condition. The vertical scale displays (VSD) change color only once from green to blue when at maximum limit and the color change is hard to detect in daylight conditions. The difficulty in quickly reading and interpreting the engine and transmission VSD's is a shortcoming.

Turbine Gas Temperature (TGT)/Gas Producer Speed (Ng)

49. The TGT indicator located on the instrument panel is 14 in. above the Ng gauge which is located on the multiparameter display (MPD) as shown in figure 2. The separation of the two gauges represents a focal length change of approximately 5.5 inch from the pilot's eye position and causes the pilot to make a slight head movement in conjunction with eye movement to refocus which prevents the pilot from properly monitoring the two gauges. During engine starts, the pilot is required to closely monitor both instruments requiring unnecessary head and eye movement. The excessive separation between the TGT indicator and the Ng indicator prevents proper monitoring during starts and is a shortcoming.

Transponder and ICS Panel

50. The AN/APX-100 transponder control panel and pilot's interphone communication system (ICS) box are located on the console as shown in figure 3. While sitting in a comfortable flying position with the collective down, the pilot can see neither the



1. Dual tachometer indicator
2. TGT and TRQ indicator
3. Magnetic compass
4. Slip indicator
5. Standby altimeter
6. Standby attitude indicator
7. Multiparameter display (MPD)
8. AN/APX transponder control panel
9. Pilot C-10414 (ICS) control panel
10. Standby airspeed indicator
11. Remote frequency display

Figure 3. Instrument Panel and Console

pilot's ICS box nor the transponder, as shown in photo 4, appendix F, without leaning to the left. With the collective at 50% travel, which is at approximately a cruise setting, the transponder and ICS box are even harder to read as shown in photo 5. During enroute, tactical IMC conditions, the pilot will be required to set various transponder codes and change settings on his ICS box. While performing these tasks, the pilot's head and body movements will be extremely vertigo inducing. The location of the transponder and ICS control box beneath the collective which requires large head and body movements by the pilot to read or set them is a shortcoming during visual meteorological conditions flight and a deficiency during single pilot IMC flight. The OH-58D should be restricted from single pilot IMC flight unless a trained CPO is on board.

Transponder Control Panel

51. The AN/APX-100 transponder control panel, shown in photo 11, is located on the console. The transponder codes are extremely hard to read due to the small windows on the transponder, requiring the pilot or CPO to be looking almost straight down at the panel. The difficulty in reading the transponder codes is a shortcoming.

Engine Start Switch

52. The start switch is oriented to operate left and right as shown with the guard down in photo 4 and the guard switch up in photo 6. The guard operates fore and aft. The start switch is not spring-loaded and stays on until the pilot turns it off. Flipping the guard down in a conventional manner will not turn the start switch off, although it appears that the switch is OFF. Damage to the starter-generator could result if the switch is left ON following engine start. The orientation of the start switch in an unconventional manner in relation to the guard is a shortcoming.

Instantaneous Vertical Speed Indicator

53. The instantaneous vertical speed indicator (IVSI) on the MFD get its signals from the radar altimeter. When flying over anything other than flat terrain, the IVSI is very erratic and does not portray what the aircraft is actually doing with respect to mean sea level. If the terrain is descending in the direction of flight, the IVSI will indicate that the aircraft is climbing when it is actually flying level. Since trend information can not be adequately obtained from the digital altimeter, the erratic IVSI information is a problem for IMC flight. The erratic and inappropriate IVSI display is a shortcoming.

Secondary Flight Instruments Location

54. The secondary flight instrument group contains a standby attitude indicator, airspeed indicator, altimeter, magnetic compass and slip indicator. The secondary flight instruments are designed to provide flight reference in the event of a partial or complete failure of the primary instrument system. The secondary flight instruments are located as shown in figure 2. Also, the secondary instruments must be used whenever the pilot selects a different display for the MFD (e.g., navigation or communication pages). The pilot is required to look down and to the left to see the three instruments on the pedestal and down and to the right to see the magnetic compass. The slip, mast torque and TGT indicators are located as shown in figure 2. The widely separated secondary flight instruments required the pilot to scan almost his entire half of the instrument panel. The location and separation of the secondary flight instruments is a shortcoming.

Circuit Breakers on the Overhead Console

55. There are six circuit breakers (CB) on the back row of the overhead console CB panel which are extremely difficult to pull due to their close proximity to the center post CB panel as shown in photo 7, appendix F. Two of the circuit breakers are the doppler and the attitude heading reference system circuit breakers which are required to be pulled and reset during a full master controller processor unit reset. To enable the pilot to pull and reset any of the six circuit breakers on the back rows, tie strings were installed as shown in photo 8. The close proximity of the back row of circuit breakers on the overhead console circuit breaker panel to the center post circuit breaker panel preventing easy access is a shortcoming.

Remote Frequency Display

56. The Remote Frequency Display (RFD) is mounted in the center of the instrument panel between the two MFD's as shown in figure 3. The unit serves as a frequency display for the five communication radios. Glare and certain lighting conditions make all information on the RFD almost unreadable. The hard to read RFD in certain lighting conditions is a shortcoming.

Built-In-Test Function

57. The built-in-test (BIT) function of the MPD allows performance of BIT on the vertical scale display subsystem. The BIT function of the MPD always showed one error (F1-05) prior to run-up as shown in photo 9. The F1-05 code indicates a rotor disk failure,

but was not accurate in that nothing had failed. The routine appearance of a rotor disk failure of the MPD BIT during prestart checks is a nuisance in that it delayed the start sequence during investigation and is considered a shortcoming.

Acceleration Cue

58. The acceleration cue, indicated by a circle on the MFD in the hover mode, is designed to show acceleration up to a maximum of 2.56 g. The acceleration cue did not properly indicate whether or not the aircraft was accelerating or in which direction the aircraft was accelerating. In a stabilized hover, the circle stayed in the upper half of the screen which indicated that the aircraft was accelerating forward. With the aircraft stationary on the ground, the circle was as shown in photo 10 indicating a forward acceleration of the aircraft. The incorrect acceleration information in the hover mode is a shortcoming.

RELIABILITY AND MAINTAINABILITY

59. The reliability and maintainability features of the OH-58D aircraft were evaluated throughout the test. Twelve equipment performance reports (EPR) were submitted during the evaluation and are listed in appendix G. The most significant reliability and maintainability problems encountered are summarized in the paragraphs below.

60. During the evaluation, some of the problems identified in EPR's were corrected. Revisions to the computer software corrected the erroneous activations of the HIGH RPM WARNING and MAST OVERTORQUE WARNING. Computer software revisions also reduced the number of PITCH/ROLL SCAS DISENGAGEMENTS and MCPU RESETS.

61. During the first half of the test program, the engine installed in the test aircraft had intermittent starting problems. The failures occurred primarily on first starts of the day or after the aircraft had been sitting for awhile. Approximately seven starts were aborted during the test program, and in all cases, a second start attempt resulted in a normal start sequence. Numerous components have been replaced and the start problem has not resurfaced. However, the reason for the engine start problem has still not been identified. Of the five preproduction OH-58D's being flown, only the aircraft being evaluated in this PAE is experiencing engine start problems.

62. The four pitch change links contacted the engine cowling while evaluating slope landings of 10 deg with the left skid

upslope. Damage to the pitch change links and cowling was as shown in photos 12 and 13, appendix F. The pitch change links were repaired and the cowling opening was enlarged. The enlarged opening in the cowling permitted 10 deg left skid upslope landings without pitch change link contact.

AIRCRAFT PITOT-STATIC SYSTEM

63. The ship's pitot-static system was calibrated in level flight and dives using a trailing bomb as a calibrated airspeed reference. The data are presented in figure 62, appendix E. The pitot-static system of the OH-58D is satisfactory.

CONCLUSIONS

GENERAL

64. The following general conclusions were reached:

a. The OH-58D has the potential to be a significant improvement over the OH-58C.

b. The high pilot workload associated with aircraft flying qualities and pilot operation of systems and displays make the OH-58D unsuitable for single pilot instrument meteorological conditions flight (ref para 31).

c. The vibrations of the OH-58D are satisfactory (ref para 42).

ENHANCING CHARACTERISTICS

65. The following enhancing characteristics were identified:

a. Changing displays on the pilot's multifunction display could be accomplished without removing either hand from the flight controls (para 44).

b. Changing preprogrammed radio frequencies could be accomplished by the pilot without removing either hand from the flight controls.

DEFICIENCIES

66. The following deficiencies were identified and are listed in order of importance:

a. Restricted cyclic control travel due to Pilot/CPO cyclic stick misalignment (para 47).

b. The location of the warning display on the multifunction display blocks out the altitude display (para 46).

SHORTCOMINGS

67. The following shortcomings were identified and are listed in order of importance:

a. The difficulty in maintaining directional trim when making small changes in flight conditions (para 29);

b. The high pilot workload encountered in left sideward flight between 15 and 25 KTAS (para 28);

c. The pitch instability in steady turns of 45 degrees bank angle or greater resulting in high pilot workload (para 16);

d. The imbalanced longitudinal cyclic forces in forward flight with hydraulics off (para 36);

e. The location of the transponder and ICS control box beneath the collective which requires large head and body movements by the pilot to read or set them (para 50);

f. The lateral control response characteristics resulting in unpredictable roll attitude control (para 24);

g. The longitudinal control response characteristics resulting in unpredictable pitch attitude control (para 23);

h. The location and separation of the secondary flight instruments (para 54);

i. The short collective delay time allowable, following a sudden engine failure, in order to remain above the minimum rotor speed of 310 RPM (para 32);

j. The difficulty in quickly reading and interpreting the engine and transmission vertical strip displays (para 48);

k. The difficulty in reading the transponder codes (para 51);

l. The divergent long-term dynamic response characteristics at 50 KCAS (para 20);

m. The erratic and inappropriate instantaneous vertical speed indicator display (para 53);

n. The hard to read Remote Frequency Display in certain lighting conditions (para 56);

o. The close proximity of the back row of circuit breakers on the overhead console to the center post circuit breaker panel preventing easy access (para 55);

p. The orientation of the start switch in an unconventional manner in relation to the switch guard (para 52);

q. The excessive separation of the turbine gas temperature indicator and the N_g indicator prevents proper monitoring during engine starts (para 49);

r. The incorrect acceleration information in the hover mode on the MFD (para 58);

s. The routine appearance of a rotor disk failure on the muitiparameter display built-in test during prestart checks (para 57).

SPECIFICATION NONCOMPLIANCE

68. The following specification noncompliance's were identified:

a. Longitudinal and lateral control response did not meet the requirements of the system specification, para 3.2.1.4.2, in that the plot at control sensitivity versus rate damping did not fall within the plot area required (para 25).

b. Slope landing characteristics did not meet the requirements of the system specification, para 3.2.1.4.5 in that nose-down slope landings are limited to 5 degrees which is 5 degrees less than the requirement.

c. Low-speed flight characteristics did not meet the requirements of the system specification, para 3.2.1.4.1, in that left sideward flight required unusual control manipulations and pilot workload was not minimal.

d. Collective delay time allowable following a sudden engine failure did not meet the requirements of the system specification, para 3.2.1.4.b, in that collective delay time of 0.9 seconds at 100 KCAS, IRP, was less than 1.2 seconds.

e. The 4 per revolution (26.32 Hz) vibration at the copilot seat did not meet the requirements of MIL-H-8501A, para 3.7.1.(b) in that the vibration level from 100 to 110 knots exceeded 0.15g but was satisfactory.

f. The OH-58D failed to meet the requirements of paragraph 3.1.7.1 in that it is unsuitable for instrument flight with only one pilot (para 31).

RECOMMENDATIONS

69. Incorporate the enhancing characteristics listed in paragraph 61 into future designs.

70. Correct the deficiencies listed in paragraph 66.

71. Correct the shortcoming listed in paragraph 63.

72. The maximum airspeed for SCAS OFF flight should be 80 KCAS.

73. The following warning should be placed in the operator's manual (para 47):

WARNING

Engaging or disengaging the CPO cyclic stick is prohibited in flight, except during emergency conditions, due to possible misalignment of the pilot/CPO cyclic controls which will severely restrict cyclic movement. During an emergency which requires the CPO stick to be engaged, the pilot's cyclic should be returned as close as possible to the neutral position before engaging the CPO cyclic stick.

APPENDIX A. REFERENCES

1. Letter, AVSCOM, DRSAV-ED, 6 February 1984, subject: Preliminary Airworthiness Evaluation of the OH-58D (AHIP).

2. System Specification, AV-SS-NTSH-B10000, Bell Model 406 *Near Term Scout Helicopter,* 24 July 1981.

3. Preliminary Draft Operator's Manual, TM55-1520-248-10, *OH-58D Helicopter,* 1 March 1984, with change 6 dated 18 September 1984.

4. Model Specificaton, Detroit Diesel Allison Division of General Motors Corporation, No. 907, *Military Turboshaft Engine, Model 250-C30R,* 17 July 1981.

5. Test Plan, USAAEFA Project No. 83-26, *Preliminary Airworthiness Evaluation of the OH-58D Helicopter,* January 1984.

6. Letter, AVSCOM, DRSAV-ED, May 1984, subject: Airworthiness Release for Preliminary Airworthiness Evaluation of the OH-58D.

7. Military Specification, MIL-H-8501A, *Helicopter Flying and Ground Handling Qualities; General Requirements for,* 7 September 1961, with Amendment 1, 3 April 1962.

8. Naval Test Pilot School Flight Test Manual, Naval Air Test Center, USNTPS FTM No. 101, *Helicopter Stability and Control,* June 1968.

9. Engineering Design Handbook, Headquarters, US Army Material Command, AMCP 706-204, *Helicopter Performance Testing,* August 1974.

10. Final Report, USAAEFA Project No. 76-11-2, *Airworthiness and Flight Characteristics Evaluation OH-58C Interim Scout Helicopter,* April 1979.

APPENDIX B. DESCRIPTION

GENERAL

1. The test helicopter is a preproduction OH-58D, built by Bell Helicopter Textron Incorporated (BHTI) S/N 69-16285. The OH-58D shown in figures 1 through 3 is an extensively modified OH-58A helicopter, incorporating major revisions in the engine, flight control system, and instrument panel. General dimensions are shown in figure 4. The OH-58D helicopter is shown in photos 1 through 8 in the primary mission configuration and shown in photos 9 through 16 in the low drag configuration.

2. Maximum takeoff gross weight of the OH-58D is 4500 pounds (4600 pounds for test purposes) compared to 3200 pounds for the OH-58C. Primary mission gross weight is 4045 pounds and the alternate mission gross weight is 4150 pounds. Major modifications to the OH-58A include:

a. a single T703-AD-700 (Allison 250-C30R) turbine engine shown in figures 5 and 6 and rated at 650 shaft horsepower (shp), sea level, and standard day;

b. a 4-bladed, foldable main rotor system;

c. a three-axis electronic stability and control augmentation system;

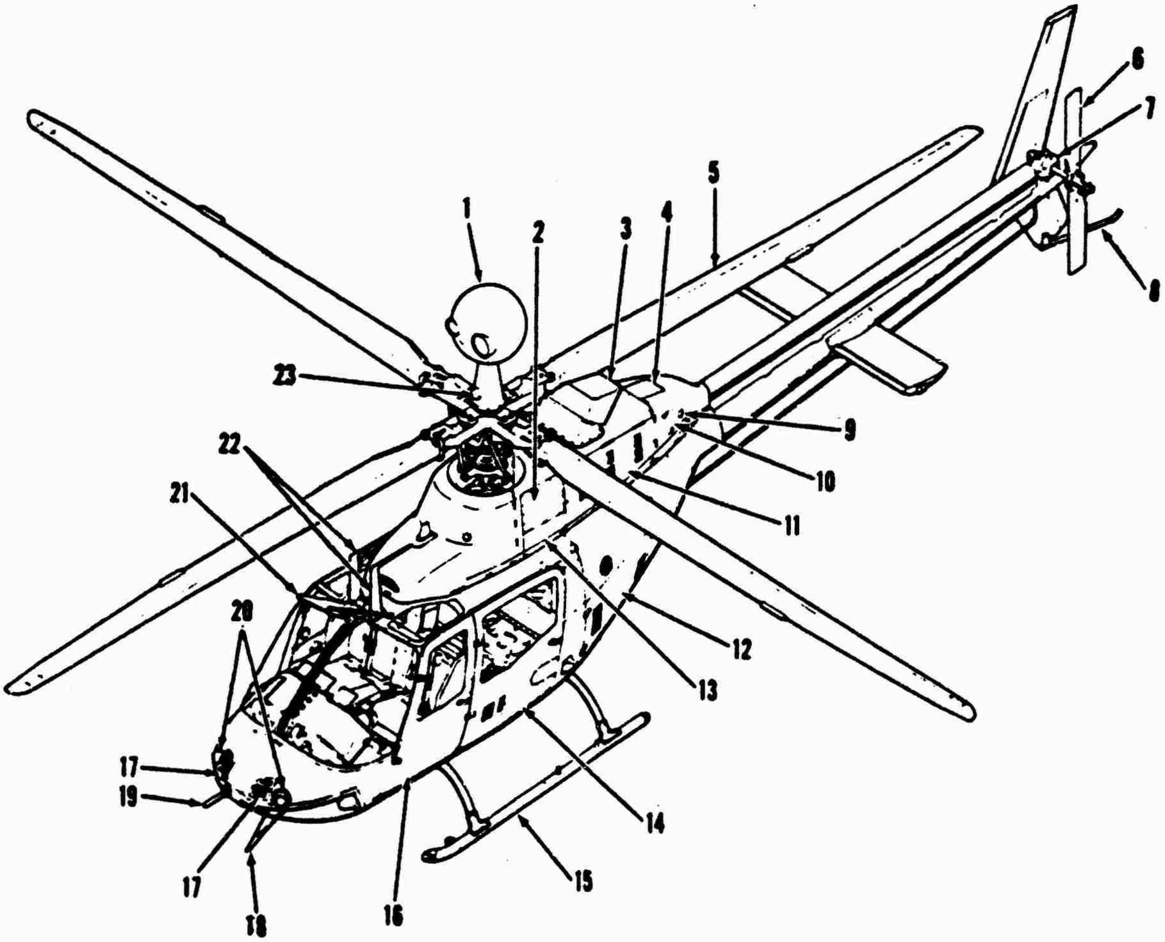
d. an extensively modified cockpit arrangement shown in figure 7 to include multifunction displays and a multifunction keyboard for display control;

e. an electro-optical visionics system (mast mounted sight) shown in photo 17 mounted above the main rotor providing day/night infrared target acquistion designation; and

f. usable fuel capacity of 104 U.S. gallons.

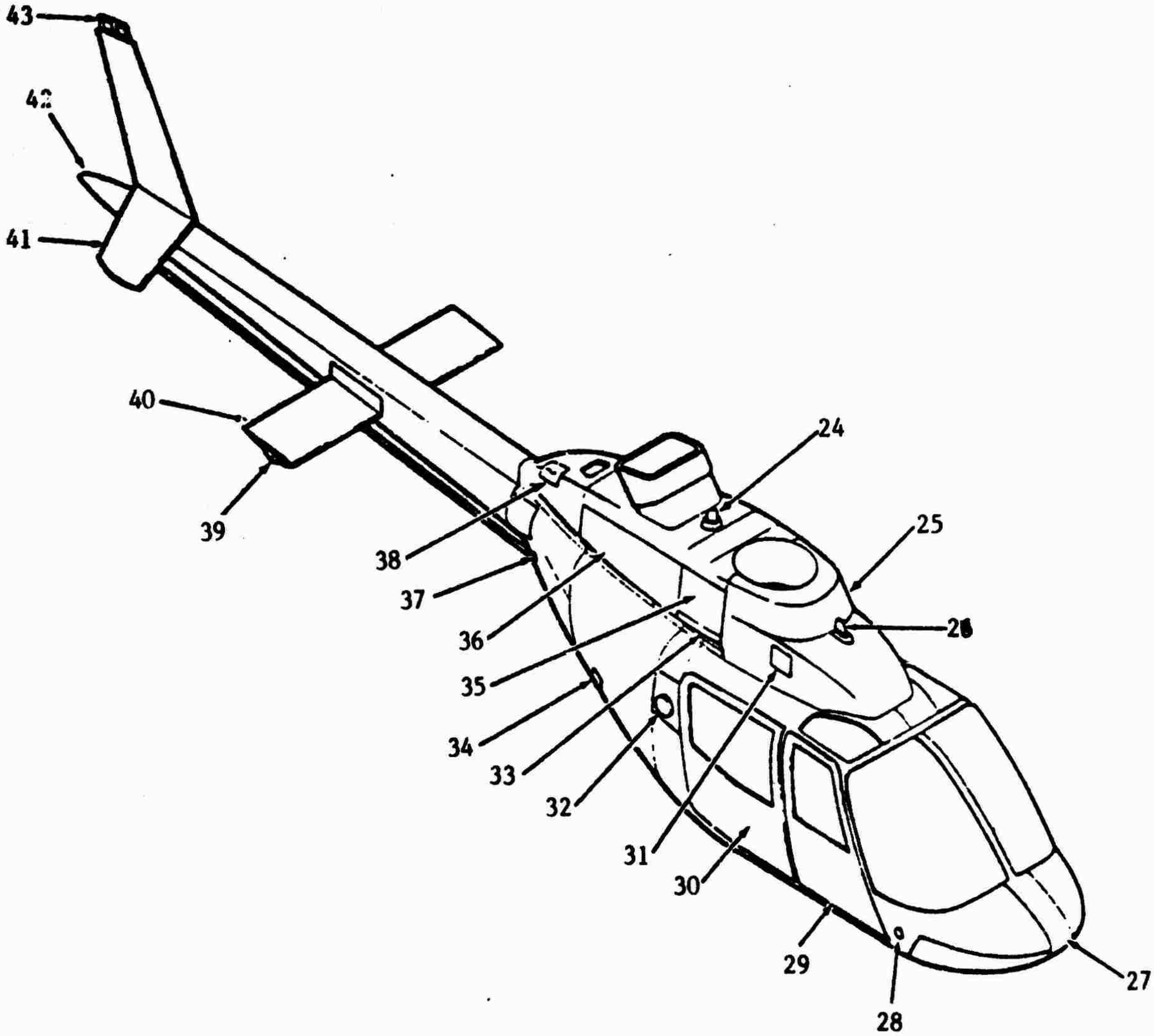
FUSELAGE

3. The fuselage consists of three main sections: the forward (or cockpit) section, the intermediate (or equipment) section, and the tail boom section. The forward section provides the cockpit, fuel cell enclosure, forward equipment compartment bay, and pylon support. The intermediate section supports the engine and oil cooler subsystem and provides the aft equipment compartment. The tail boom section supports the horizontal stabilizer, the vertical stabilizer, the tail rotor drive system, the tail rotor and the tail rotor controls.



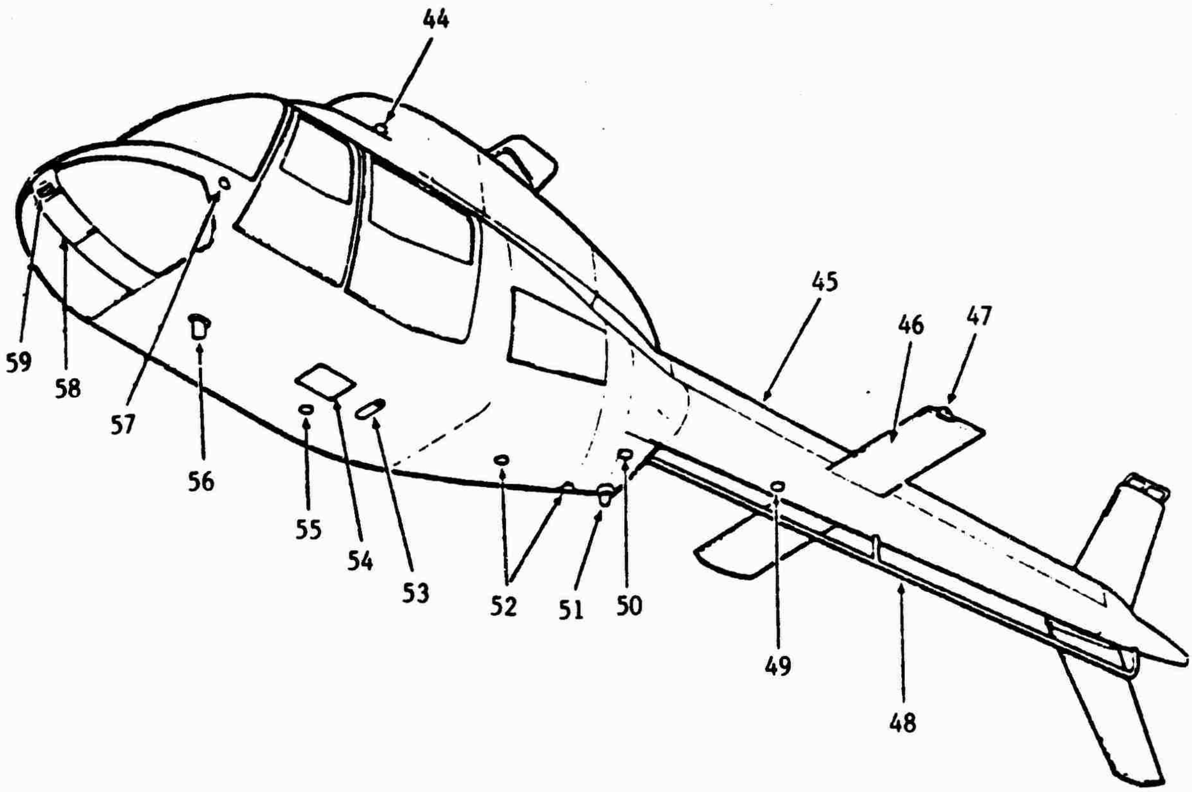
1. Mast mounted sight
2. Engine inlet
3. Engine exhaust
4. Oil cooler fan exhaust
5. Main rotor blades
6. Tail rotor blades
7. Tail rotor gearbox
8. Tail skid
9. Engine oil reservoir sight glass
10. Oil tank compartment access door
11. Left engine access door
12. Aft electrical compartment
13. Left transmission access door
14. Left cabin door
15. Landing gear
16. Left crew door
17. Ram air grille
18. Lower wire cutter
19. Pitot tube
20. Front radar warning antennae
21. Upper wire cutter
22. Secondary FM (FM homing) antennae
23. Turret cooling fan screen

Figure 1. General Arrangement



24. Upper anti-collision light
25. Transmission cowling
26. IFF antennae
27. Battery acċess door
28. Right static port
29. Right crew door
30. Right cabin door
31. Hydraulic servo access door
32. Fuel filler cap
33. Right transmission access door
34. AC external power receptacle
35. Engine inlet
36. Right engine access door
37. Right aft radar warning antennae
38. Engine oil filler access door
39. Right position light
40. Horizontal stabilizer
41. Vertical fin
42. Aft position light
43. VHF FM/VHF AM antennae

Figure 2. General Arrangement



44. Hydraulic reservoir sight glass
45. Tail rotor driveshaft cover
46. Horizontal stabilizer
47. Left position light
48. HF antennae
49. IFF antennae
50. Left aft radar warning antennae
51. Lower anti-collision light
52. Radar altimeter antennae
53. Fuel tank sump drain
54. Doppler antennae
55. Radar warning antennae
56. UHF antennae
57. Left static port
58. Search light
59. DC external power receptacle

Figure 3. General Arrangement

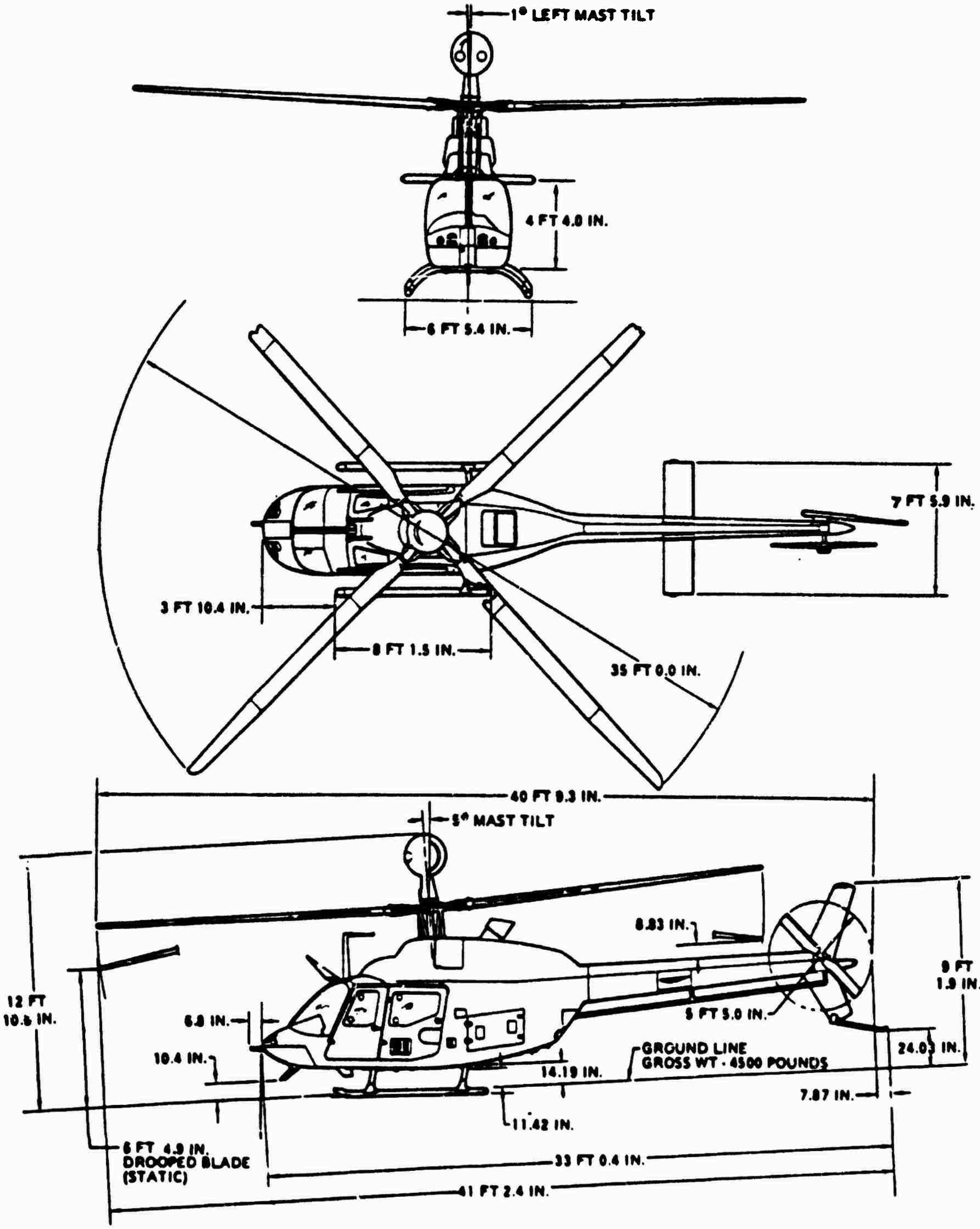


Figure 4. Principal Dimensions and Ground Clearance

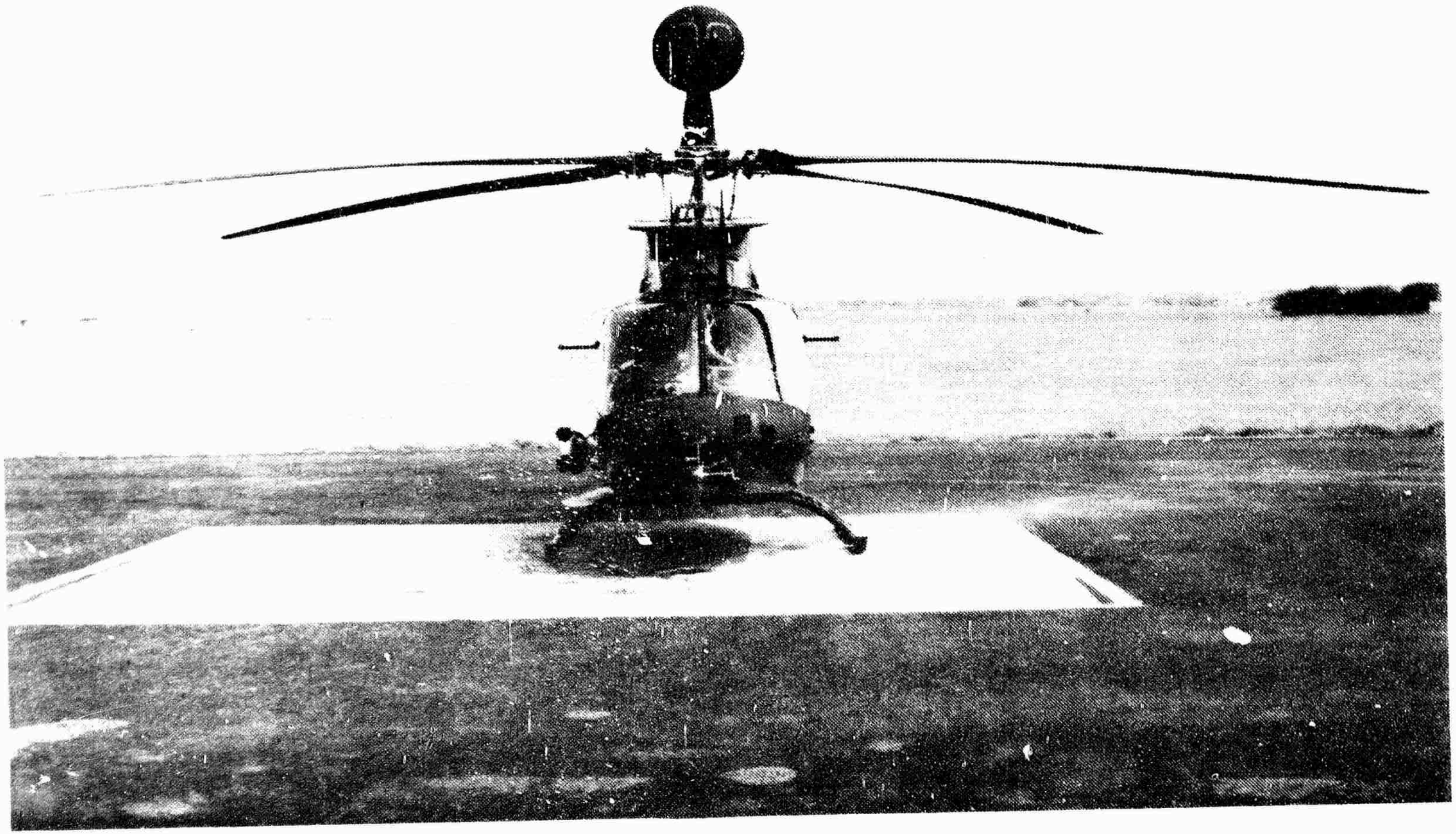


Photo 1. Front View (Primary Mission Configuration)

Photo 2. Left Front Quartering View

Photo 3. Left View

Photo 4. Left Rear Quartering View

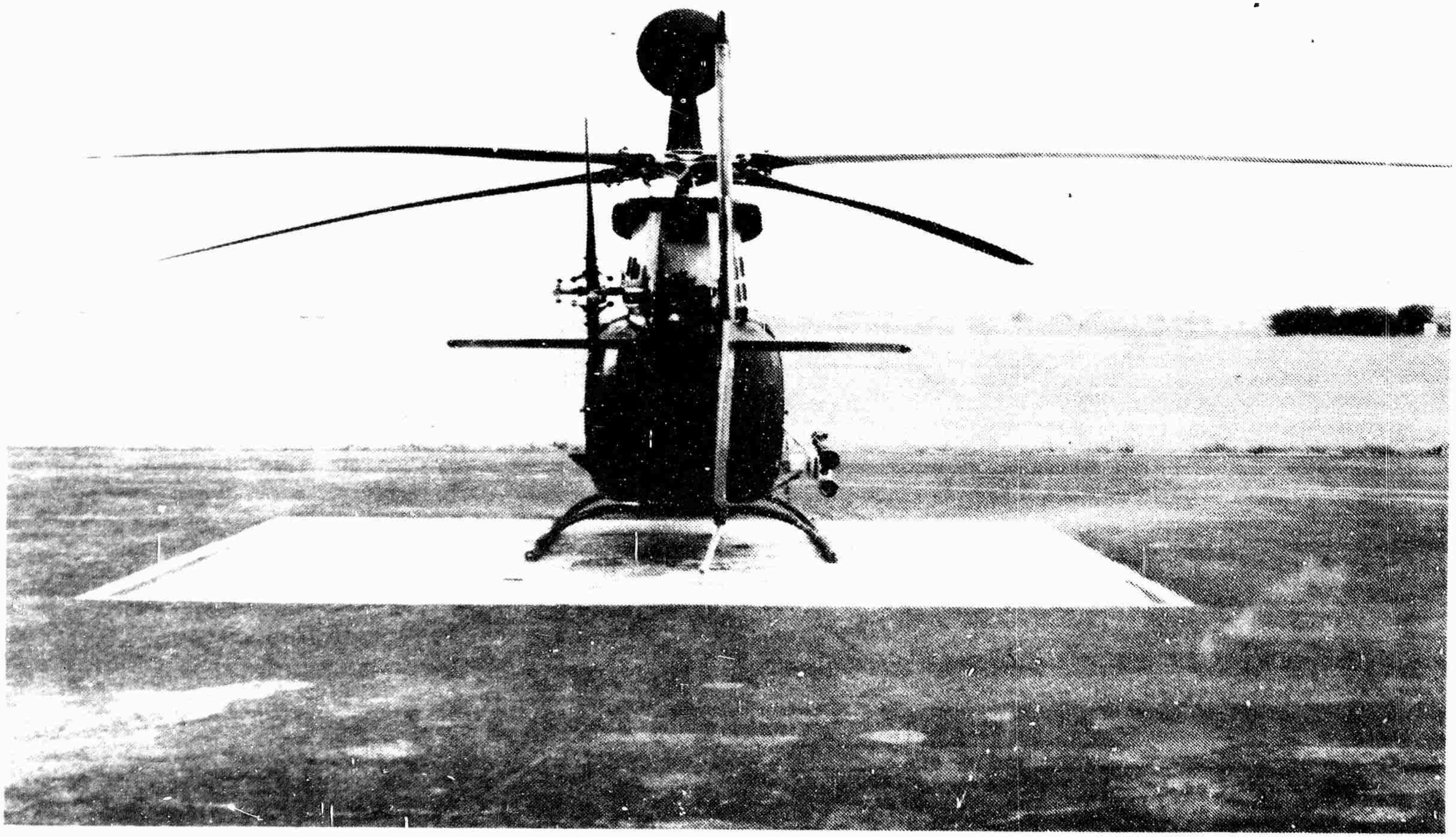


Photo 5. Rear View

Photo 6. Right Rear Quartering View

Photo 7. Right View

Photo 8. Right Front Quartering View

Photo 9. Front View (Low Drag Configuration)


Photo 10. Left Front Quartering View

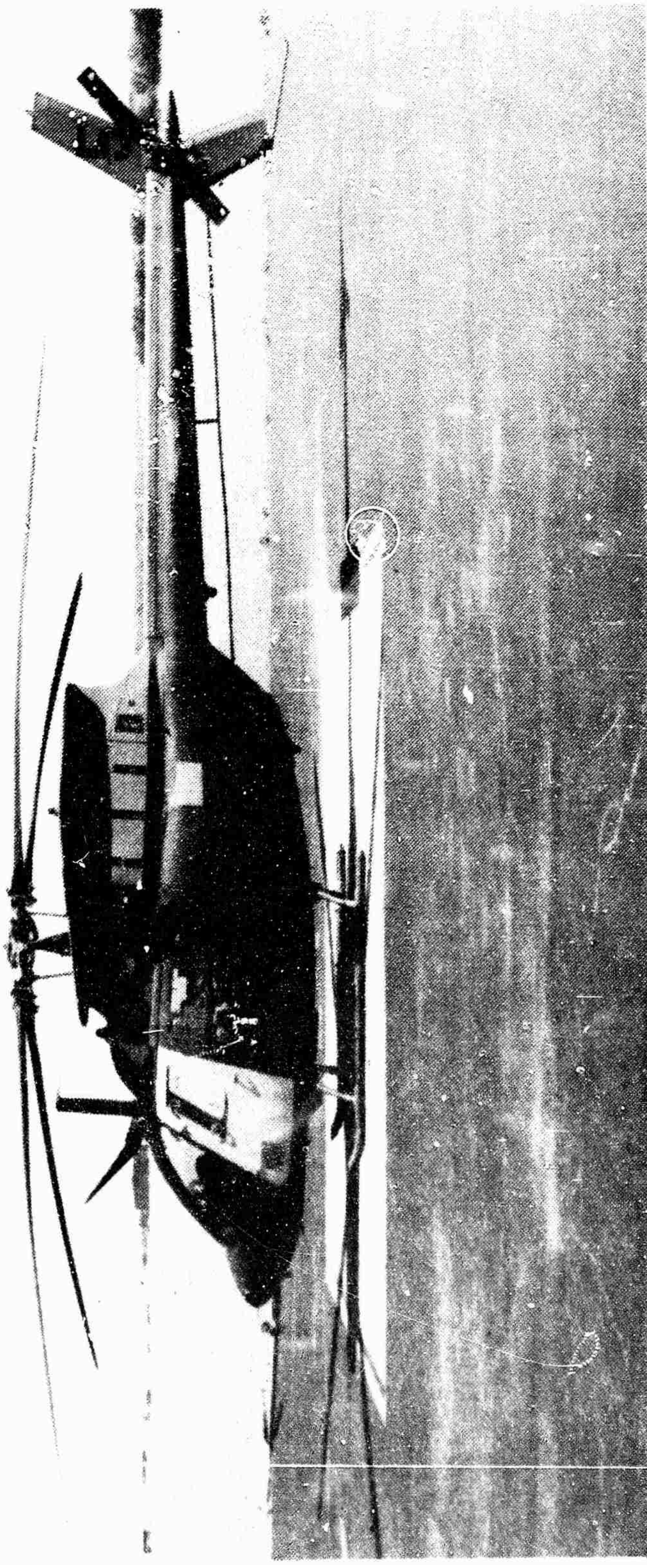


Photo 11. Left View

Photo 12. Left Rear Quartering View

Photo 13. Rear View

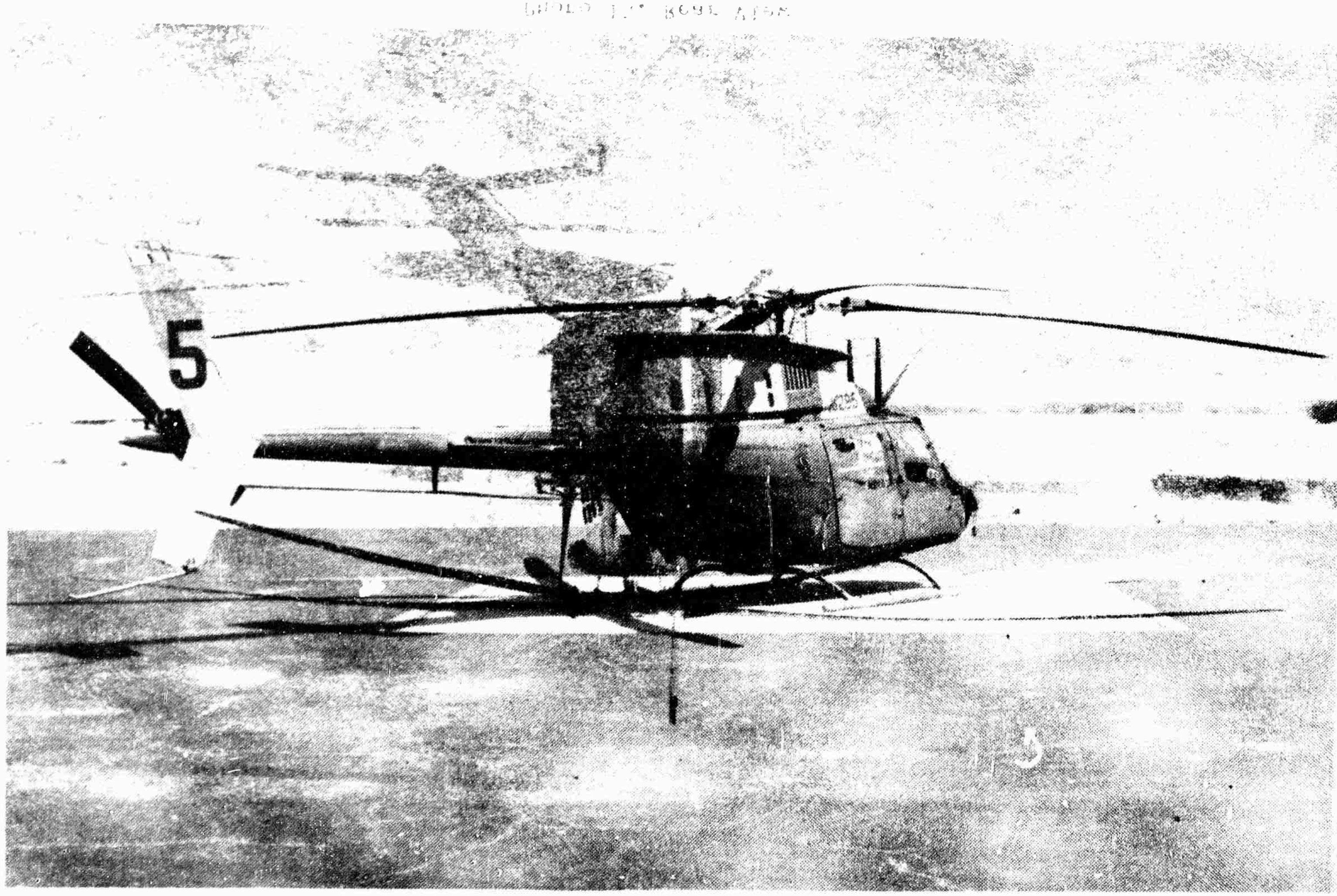
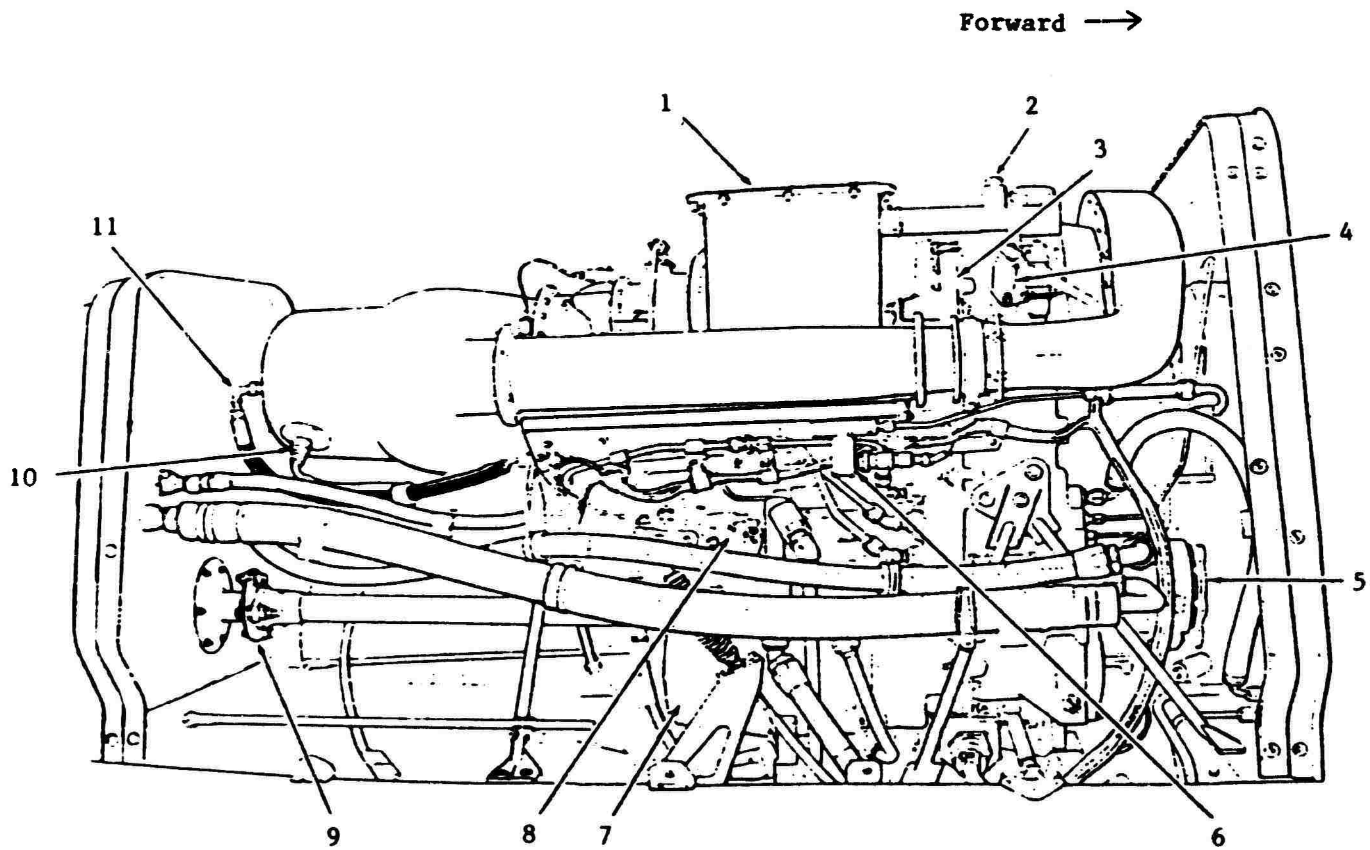


Photo 14. Right Rear Quartering View

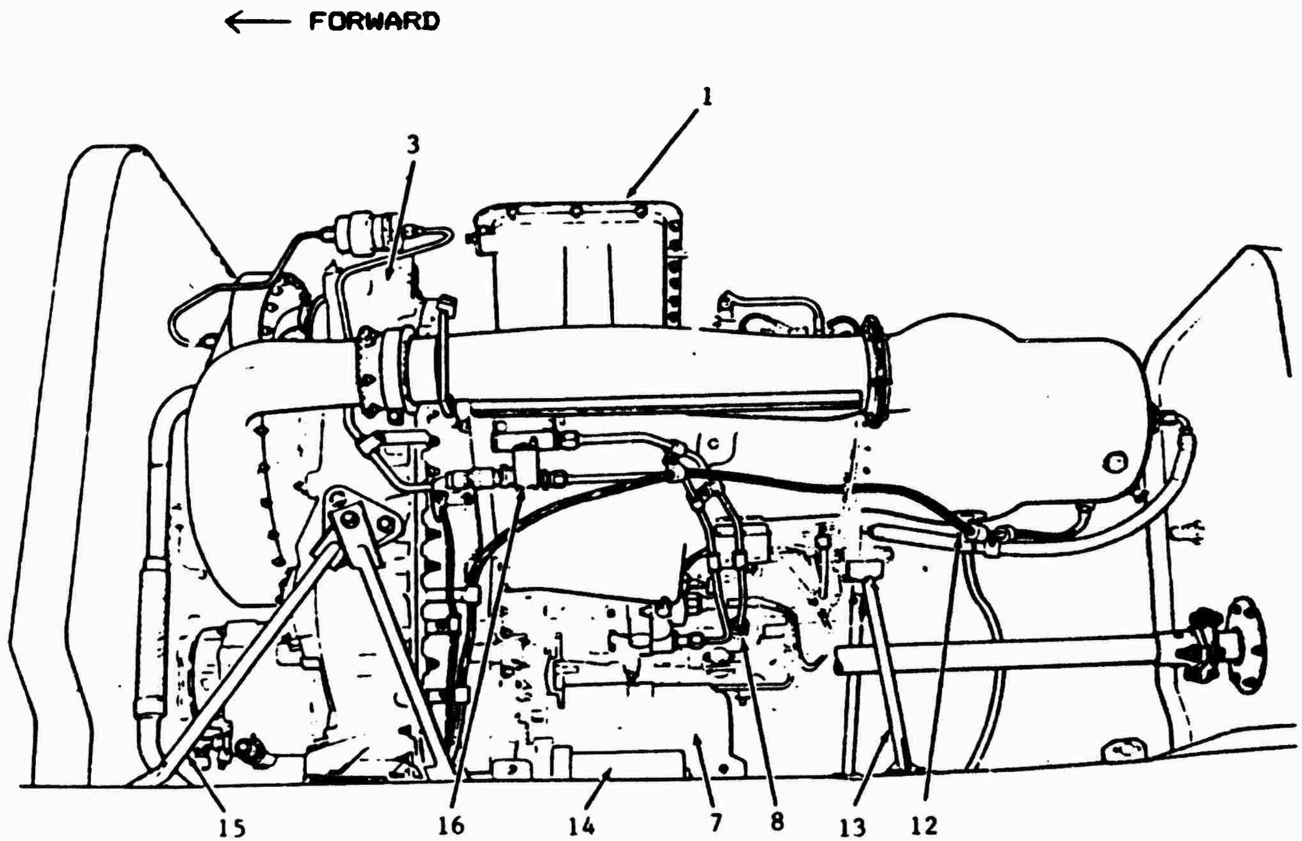
Photo 15. Right View

Photo 16. Right Front Quartering View



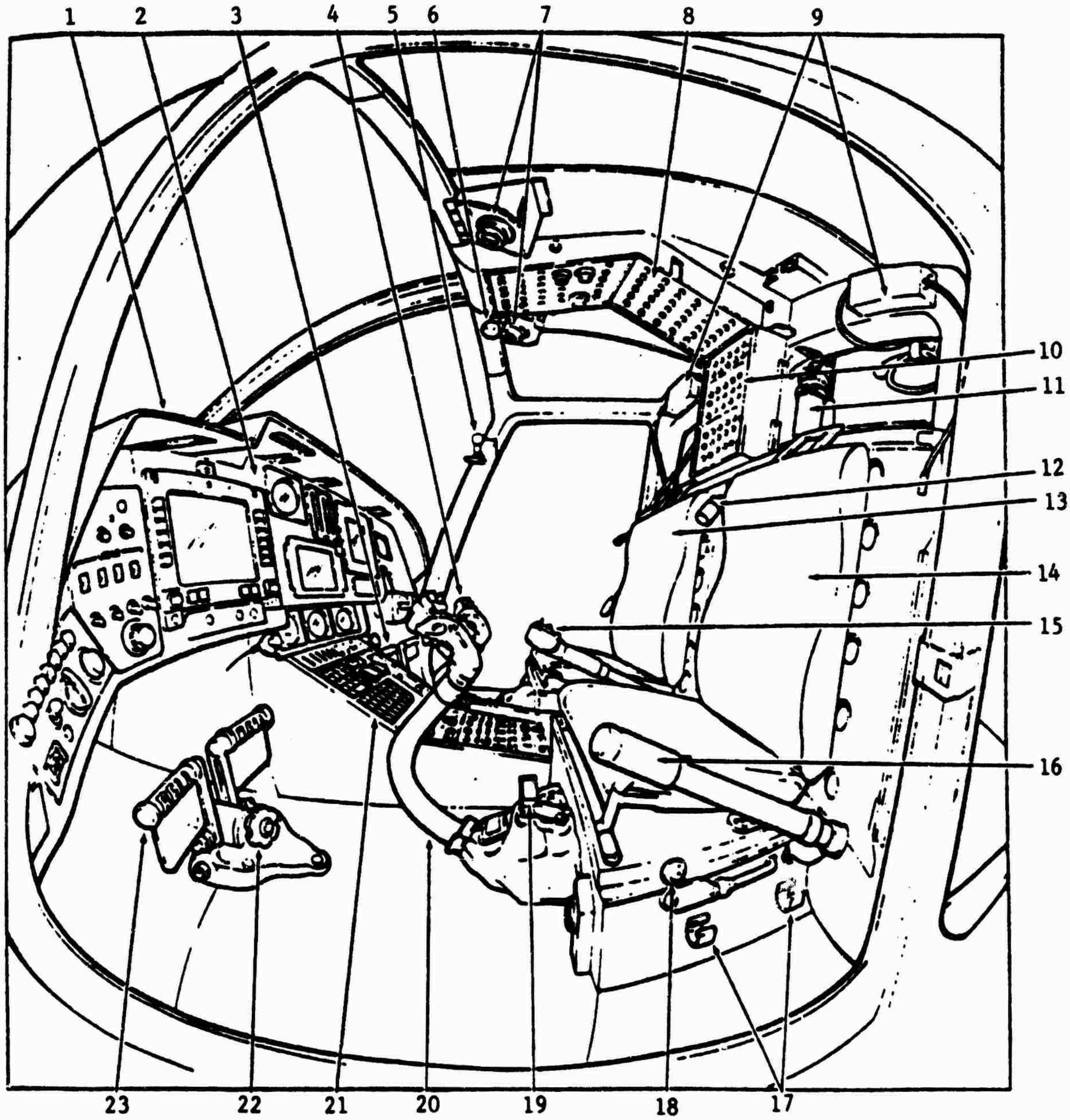
1. Exhaust
2. Oil bypass indicator
3. Power and accessories gearbox
4. Gas producer (NG) pickup unit
5. Main drive shaft
6. Anti-icing solenoid valve
7. Starter-generator
8. Fuel control unit
9. Tail rotor drive
10. Igniter plug
11. Fuel injector

Figure 5. Engine (Right Side)



12. Combustion chamber drain valve
13. Aft engine mount
14. Fuel filter
15. Forward engine mount
16. Np overspeed solenoid

Figure 6. Engine (Left Side)



1. Glareshield
2. Instrument panel
3. Pilot anti-torque pedals
4. Pilot cyclic stick
5. Pilot door jettison lever
6. Fuel shutoff lever
7. Flood lights
8. Overhead console
9. NVG, power supply
10. Center post console
11. Fire extinguisher
12. Utility light
13. Pilot seat
14. CPO seat
15. Pilot collective
16. CPO collective
17. CPO collective stowage rac
18. CPO shoulder harness lock
19. CPO cyclic lockout handle
20. CPO cyclic stick
21. Pedestal console
22. CPO anti-torque pedal adjustment knob
23. CPO anti-torque pedals

Figure 7. Crew Station

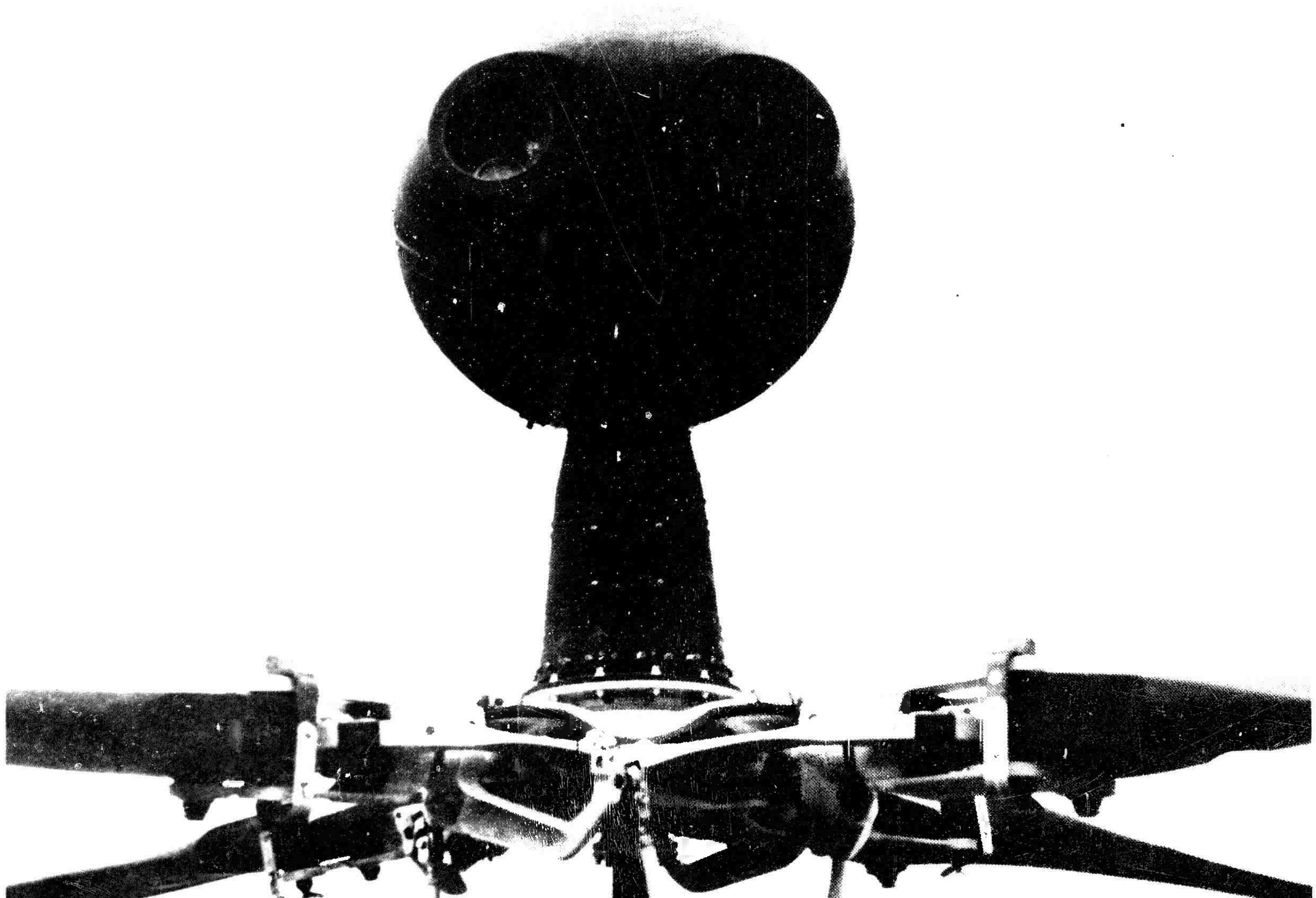


Photo 17. Electro-optical Visionics Systems (Mast Mounted Sight)

4. The basic structure of the fuselage forward section consists of a lower curved honeycomb-sandwich panel and two upper longitudinal aluminum alloy beams displaced one on each side of the fuselage. The pylon, consisting of main rotor, rotating controls and transmission is attached to the airframe via an isolation system. The pylon and isolation system are attached to the upper longitudinal beams. The upper and lower structures are interconnected by three fuselage bulkheads and a center post to form an integrated structure. The most forward and aft bulkheads act as carry-through structure for the skid landing gear cross-tubes.

5. The intermediate section supports the engine and is a conventional semi-monocoque aluminum alloy structure using the sideskins and longerons as load-carrying members. An equipment compartment provides for storage of tie-down devices, protective covers and other related helicopter equipment.

6. The tail boom section is a tapered monocoque structure with circular aluminum bulkheads and aluminum alloy skins. The tail-boom section supports the tail rotor gearbox assembly, the vertical fin, and the horizontal stabilizer. The horizontal stabilizer has been extended six inches on each side for a total span of 7 ft 5.9 inch.

ENGINE

7. The OH-58D helicopter is equipped with a turboshaft engine (shown in figures 5 and 6), model T703-AD-700, built by Allison division of Detroit Diesel Corporation. Four major components of the engine are the compressor section, the combustor section, the power turbine section, and the power and accessories gearbox. The four major systems are the engine fuel system, engine lubrication system, engine electrical system and the engine anti-icing system.

8. The compressor section is a single stage, single entry centrifugal flow compressor and is directly coupled to a two stage turbine drive. The single combustion section consists of an outer combustion case, a combustion liner, a fuel injector and one igniter plug. The power turbine section consists of a gas producer turbine rotor, a power turbine rotor, and a turbine and exhaust collective support. The aft engine mount is located on the bottom rear of the turbine support.

9. The power and accessories gearbox consists of the gas producer turbine drive gear train and the power turbine drive gear train.

All engine components, including the engine mounted accessories, are installed on the gearbox. The power and accessories gearbox incorporates two monopole pickup units to sense and control gas producer turbine and power turbine speed. Components and accessories driven by the gas producer drive train are the compressor, fuel pump, fuel control unit, pressure and scavenge oil pump and the starter generator drive. The power turbine drives two spare drive pads and supplies the power output for the main drive shaft and tail rotor drive. The gear case serves as the structural support of the engine through two engine/airframe mounts on the side of the case.

POWER TRAIN SYSTEM

General

10. The power train system consists of the freewheeling assembly, main drive shaft, transmission and mast assembly, oil cooler fan assembly, tail rotor drive shaft assembly, and tail rotor gearbox as shown in figure 8. Also included are the related components such as temperature and pressure indicators, torque indicator, electro-magnetic chip detectors, oil pump, and oil filter.

Freewheeling Assembly

11. The freewheeling assembly is mounted on the lower portion of the engine power and accessories gearbox. A freewheeling mode allows free rotation of the rotor system as well as necessary accessories when power is not being applied by the engine. An electro-magnetic chip detector (fuzz burner type) is in the freewheeling assembly.

Main Drive Shaft

12. The main drive shaft transmits engine power from the freewheeling assembly to the transmission. The Kaflex input drive shaft is equipped with non-lubricated flexible couplings at each end to accomodate minor misalignments between the main transmission and the engine mounted freewheeling unit.

Transmission and Mast Assembly

13. The transmission and mast assembly transfers the engine torque to the main rotor system with a two-stage gear reduction. The main transmission is rated at 455 mast shaft horsepower (shp) for continuous operation and a maximum transient shp of 637. The main rotor mast torquemeter is designed to allow usage of

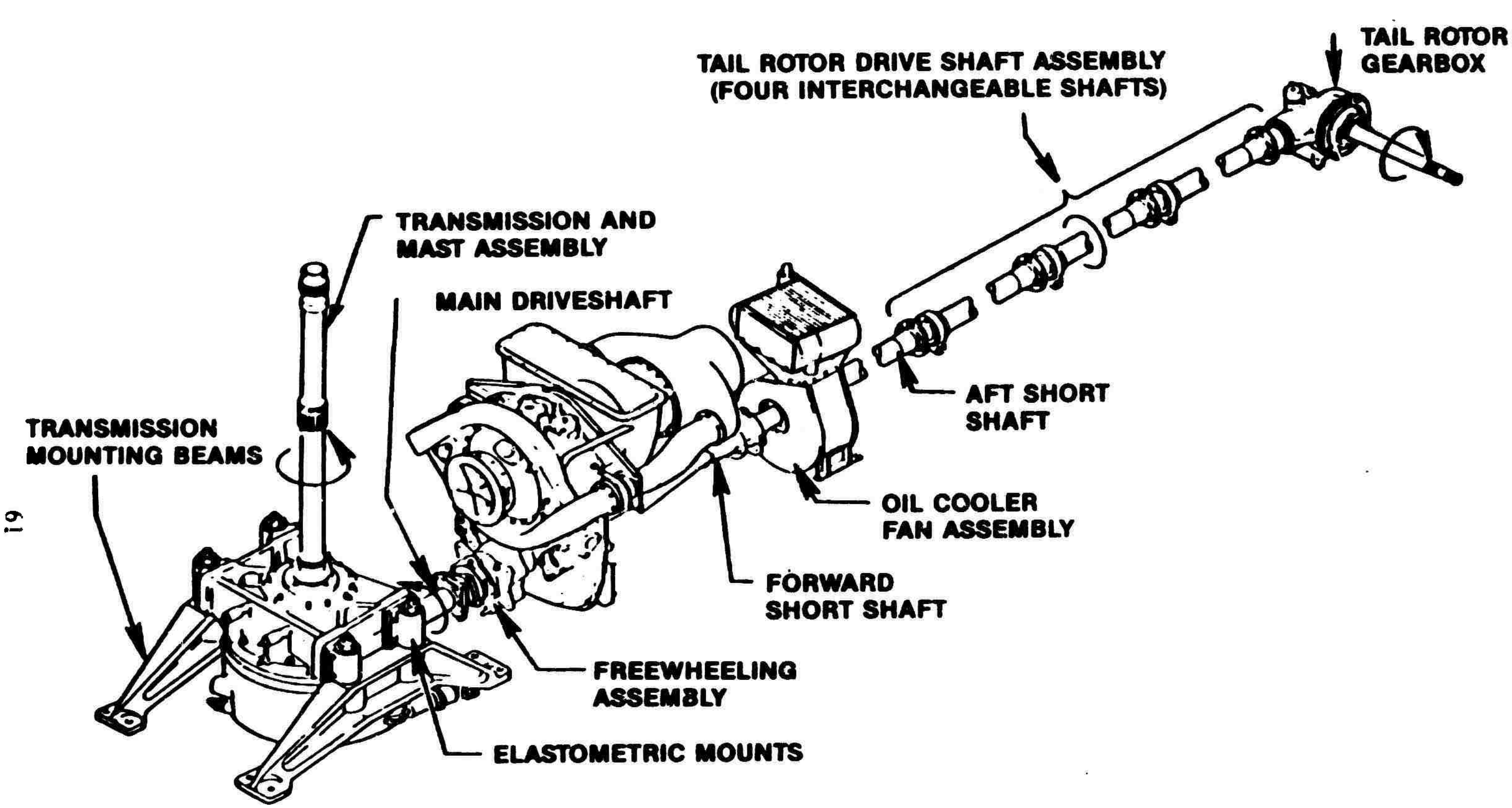
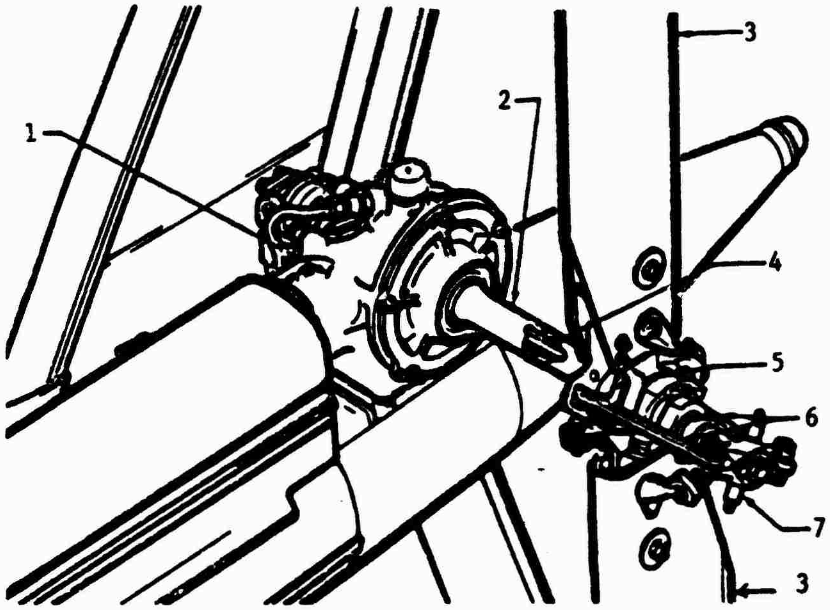


Figure 8. Power Train System



1. Tail rotor gearbox
2. Output shaft
3. Blade
4. Pitch change control tube
5. Tail rotor yoke
6. Pitch change links
7. Crosshead

Figure 9. Tail Rotor System

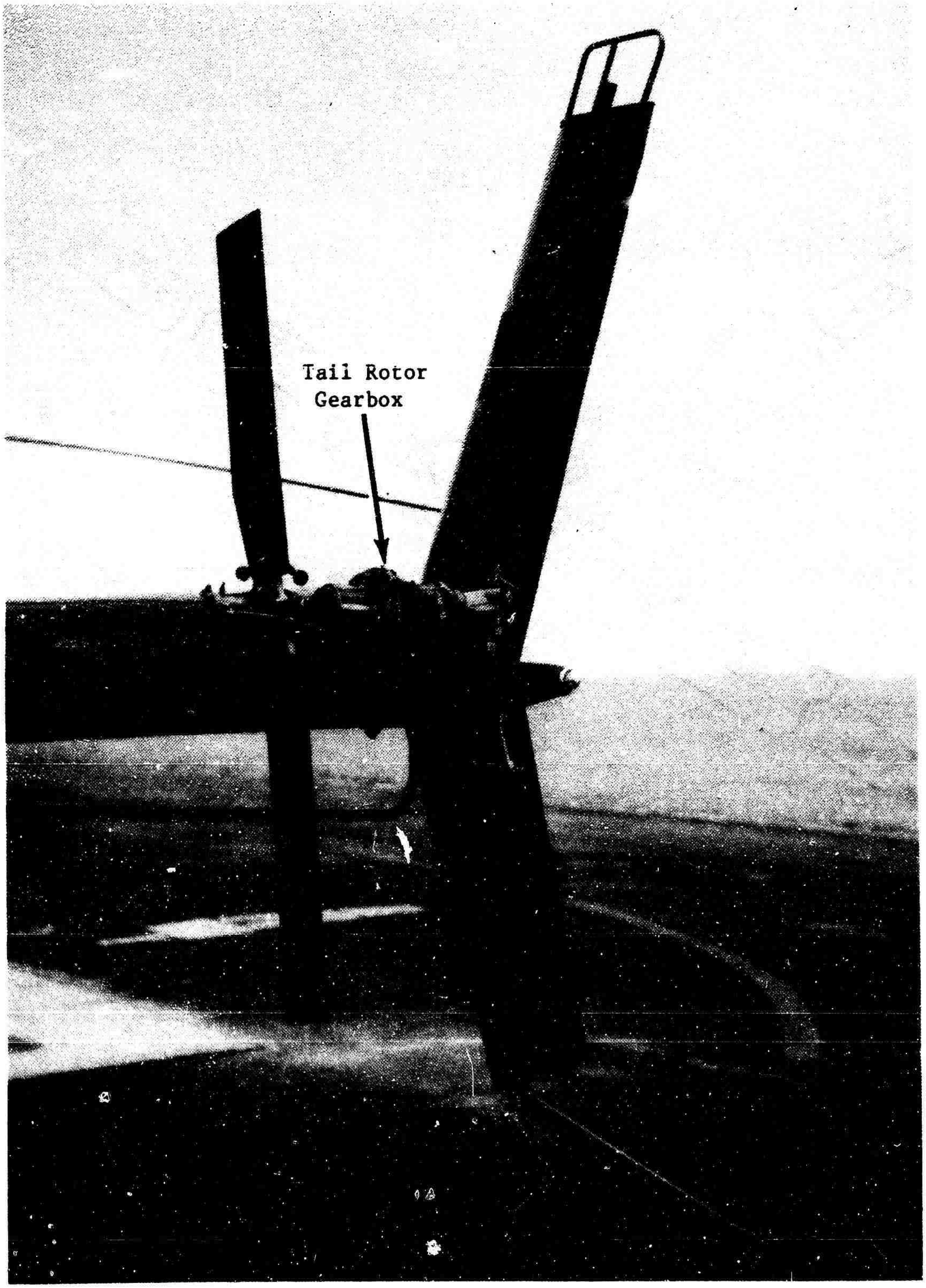
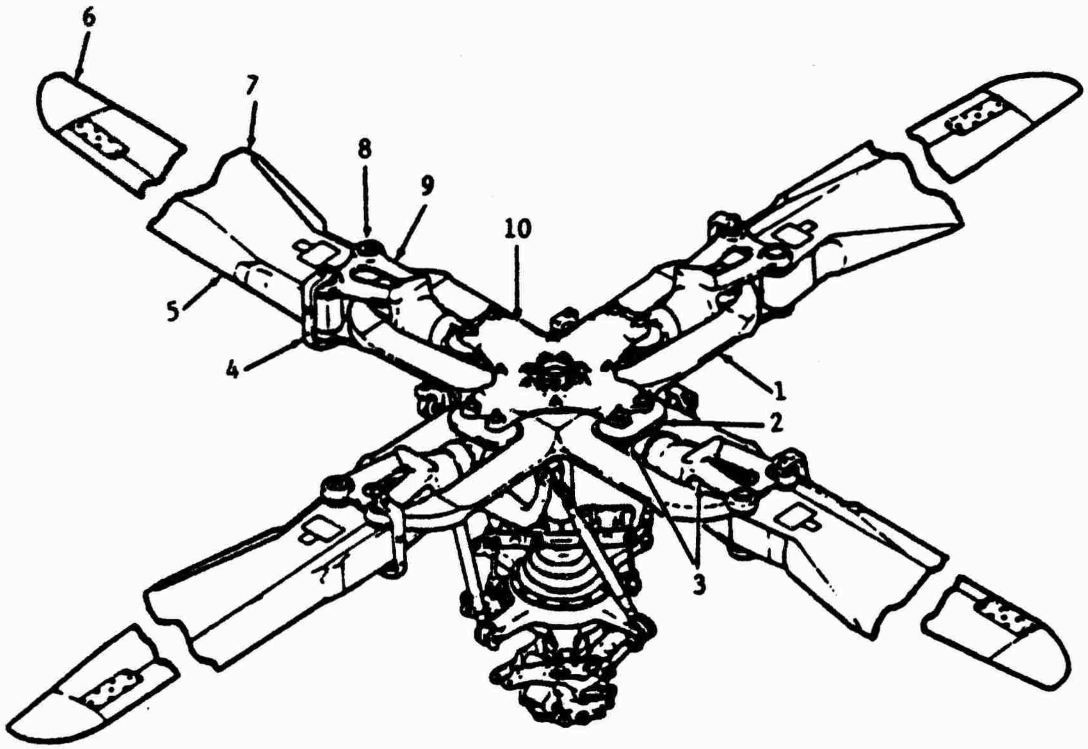


Photo 18. Tail Rotor and Vertical Fin



1. Yoke
2. Lead-lag dampers
3. Pitch change bearings
4. Pin type expandable bolt
5. Abrasive strip
6. Cap
7. Blade
8. Bolt
9. Spindle grip
10. Plate

Figure 10. Main Rotor System

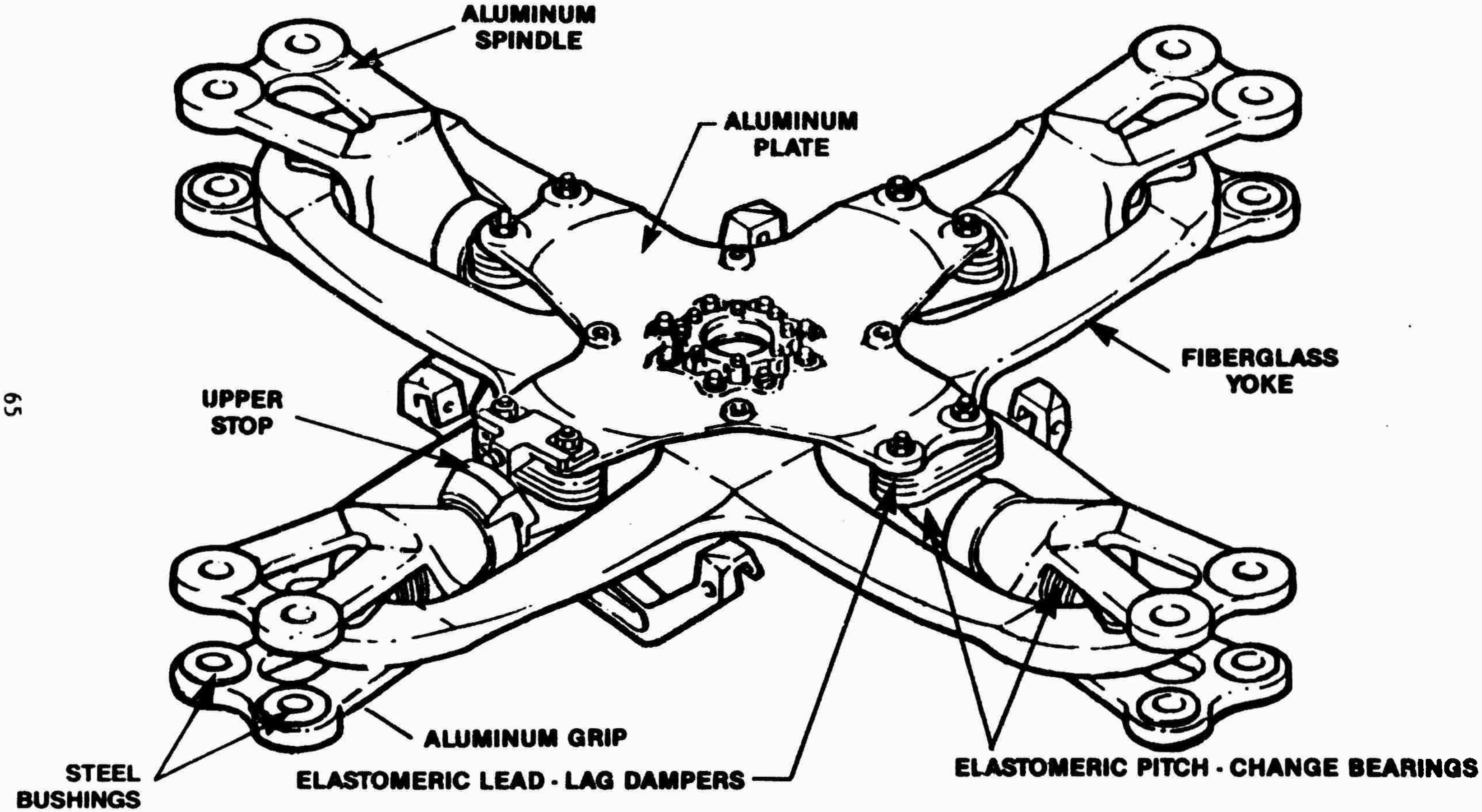


Figure 11. Composite Main Rotor Hub

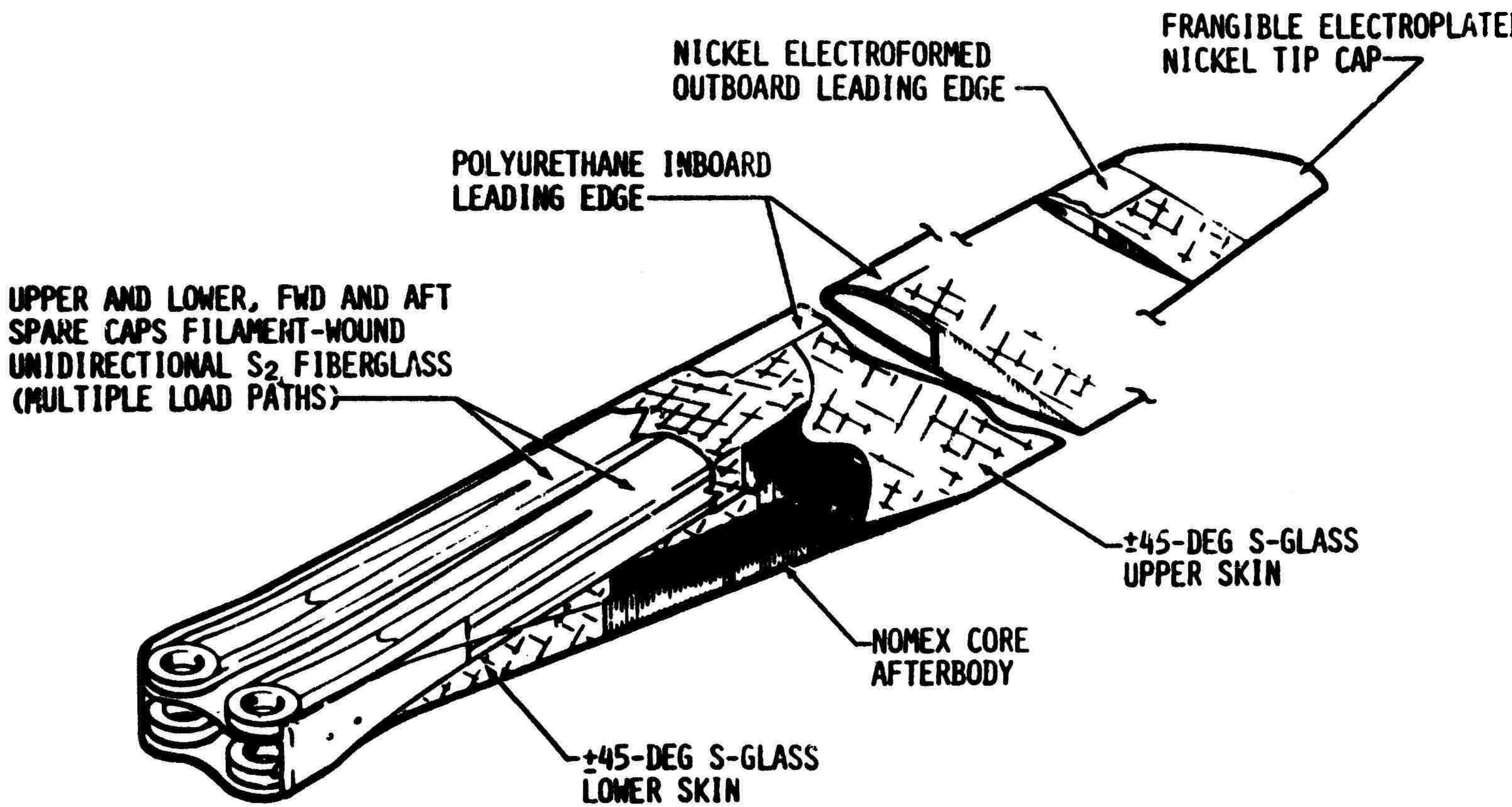
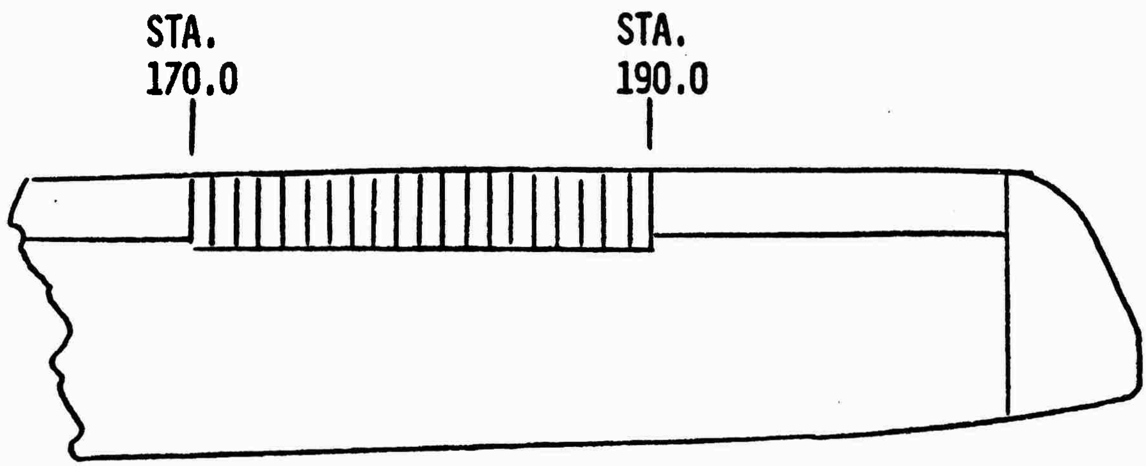


Figure 12. Fiberglass Main Rotor Blade



WEIGHT INSTALLED IN 20 ONE INCH STRIPS 6 INCHES LONG.

WEIGHT PER STRIP = 0.152 POUNDS.

Figure 13. Main Rotor Blade with External Weight

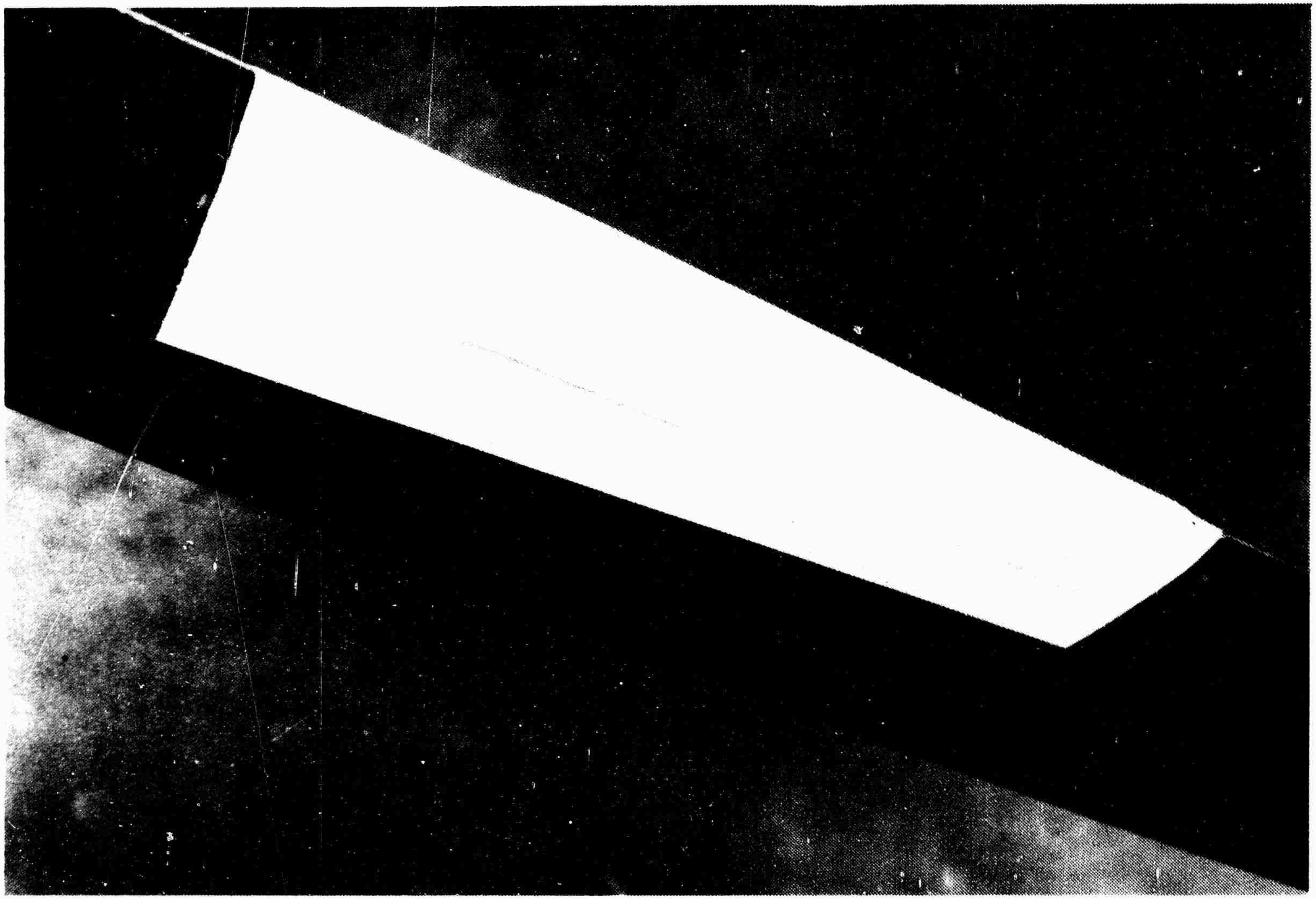


Photo 19. External Weight on Rotor Blade

full transmission rating and is mounted to the roof of the helicopter, forward of the engine, by the pylon assembly. There are two restraint spring assemblies to restrain the pitching motion of the transmission assembly. The transmission incorporates two electro-magnetic chip detectors (fuzz burner type).

Oil Cooler Fan Assembly

14. The oil cooler fan assembly is located aft of the engine. It consists of a heat exchanger/blower unit that is shared with the engine oil cooling system. The fan of this unit is attached to the tail rotor drive shaft and forces air through the heat exchanger.

Tail Rotor Drive Shaft Assembly

15. The tail rotor drive shaft assembly delivers torque from the freewheeling assembly to the tail rotor gearbox. The system consists of one steel shaft, one steel oil cooler shaft and five aluminum shafts. These shafts are connected at each end by a flexible coupling. The system is supported by six hanger bearings, each being equipped with a grease fitting for lubrication.

Tail Rotor Gearbox Assembly

16. The tail rotor gearbox assembly shown in figures 8 and 9 and photo 18 is attached to the aft end of the tailboom. The 90 degree gearbox serves as the final drive for the tail rotor. An electro-magnetic chip detector (fuzz burner type) is located on the lower right side of the casing.

ROTORS

Main Rotor

17. The main rotor system shown in figure 10 consists of four composite blades mounted to an all composite yoke shown in figure 11 incorporating elastomeric lead-lag dampers and pitch change bearings. All four blades can be folded back to aid in parking and transporting the helicopter. Each blade is attached to the grips by one bolt and one pin-type expandable bolt. The composition of the main rotor blades is shown in figure 12. An evaluation of the autorotational characteristics of the aircraft was conducted with lead weight of 3.04 pounds externally bonded to each main rotor blade as shown in figure 13 and photo 19. The weight was installed on the leading edge of the rotor in 20 strips, each of which was one inch wide and six inches long.

Tail Rotor

18. The tail rotor system shown in figures 9 and 14 and photo 18 consists of two fiberglass blades mounted to a stainless steel flexbeam yoke. Power to the system is received through the tail rotor gearbox. Pitch change inputs from the pilot or copilot anti-torque pedals are made through a pitch change control tube that slides through the center of the gearbox output shaft. A crosshead on the end of the control tube serves as the attaching points for the pitch change links.

FLIGHT CONTROL SYSTEM

General

19. The flight control system is a positive mechanical type, actuated by conventional helicopter controls. Complete controls are provided for both crewmen. The system includes a cyclic control system, a collective control system, a directional anti-torque control system, force trim system, and a stability and control augmentation system (SCAS). Hydraulic servo actuators are employed in the control system designed to prevent feedback forces and reduce pilot fatigue.

Cyclic Control System

20. The cyclic control sticks, shown in figure 15, are located forward of each crewmember seat and are the primary attitude control. The pilot cyclic stick is adjustable 1.5 inches forward and aft, or a total of three inches. It is adjusted to the desired position by turning the adjustment knob. Stick position is indicated on the indicator located at the base of the cyclic. The copilot cyclic stick can be engaged or locked out of the flight control system. When engaged, it has a typical dual control function of the flight control system. When in the lockout position, the cyclic is mechanically disconnected from the flight control system at a fixed, center position. A lever inside the cyclic dust boot must be manually repositioned to lockout the copilot cyclic stick. To reengage his cyclic, the copilot must pull up on a lever located on the lower front on the seat. All electrical switches on the cyclic grip remain functional. This feature is primarily provided for use when the MMS is being operated. The pilot and copilot cyclic grips are shown in figure 16.

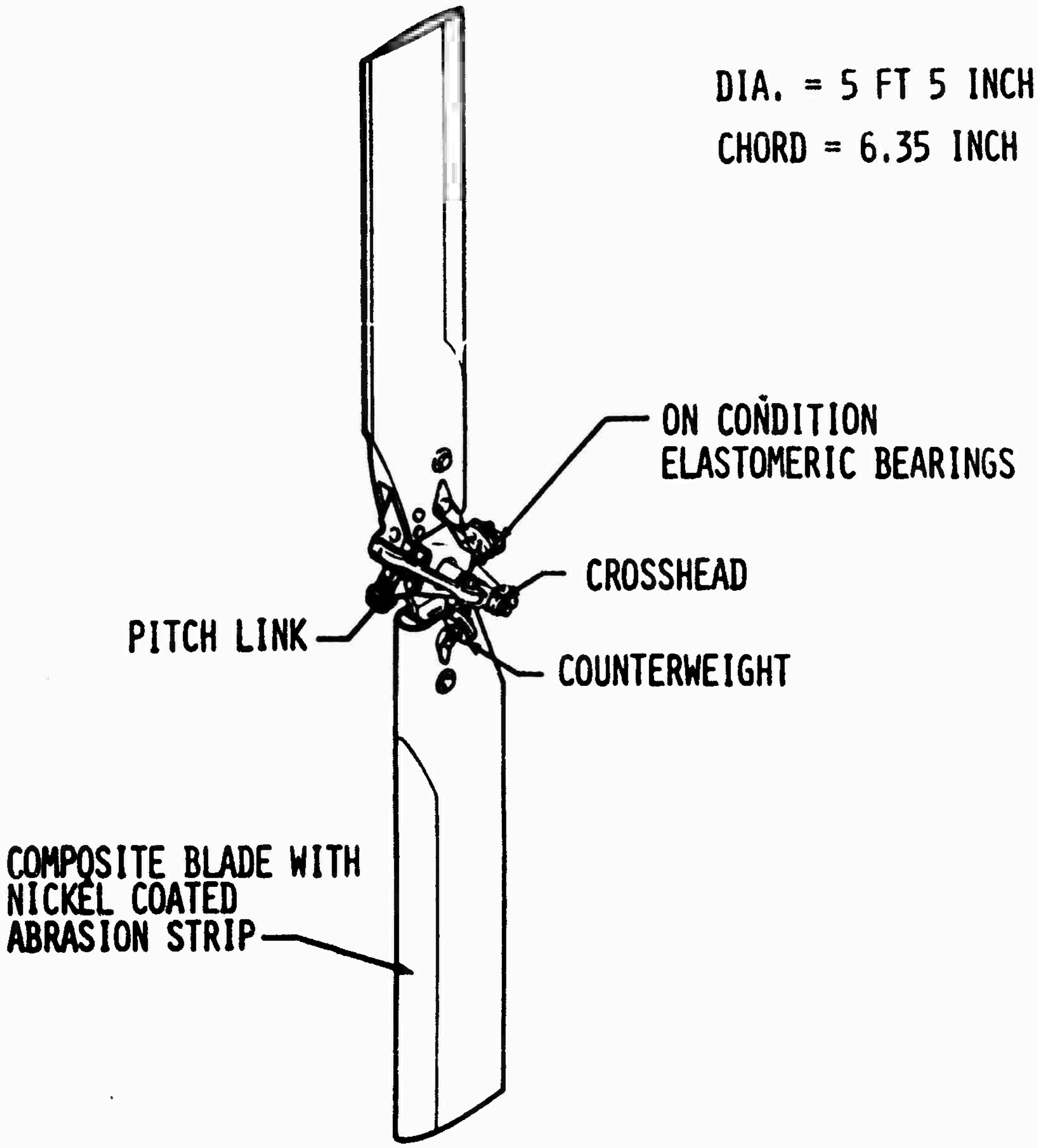
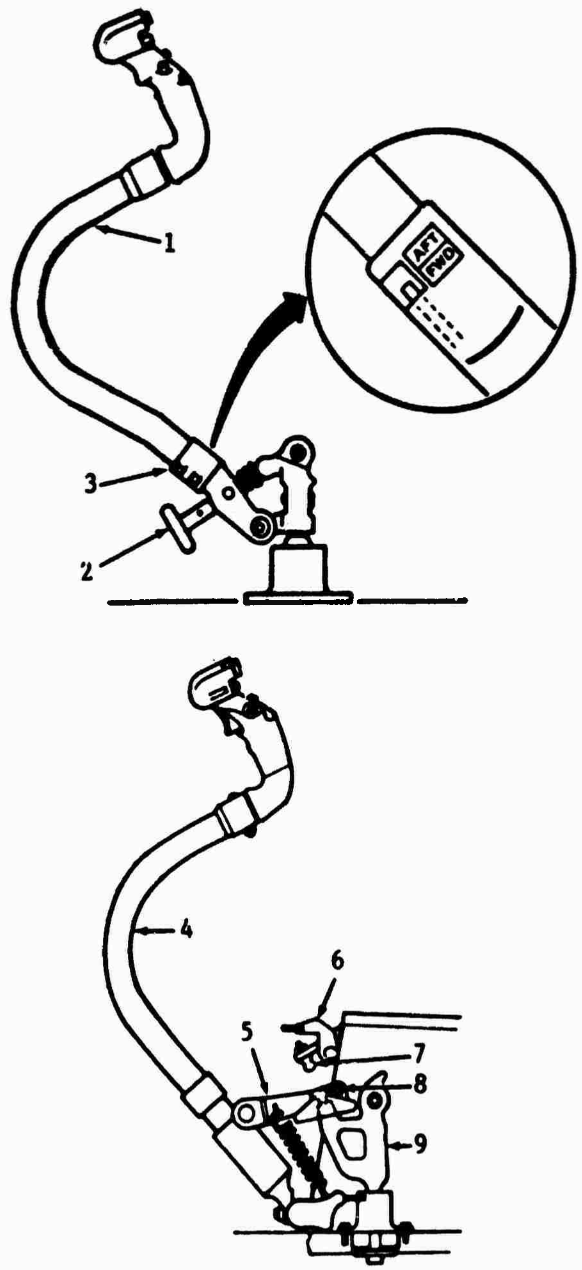


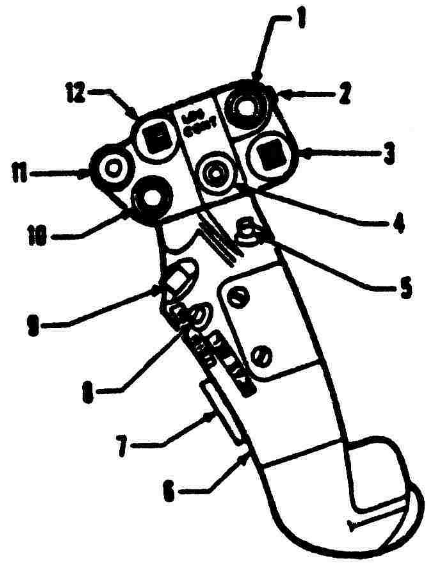
Figure 14. Tail Rotor



Engaged Position

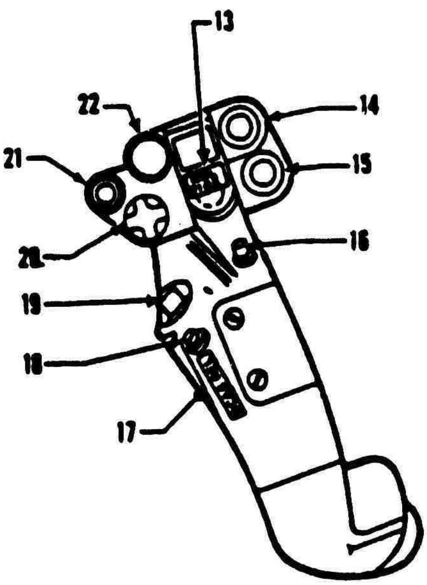
1. Pilot cyclic stick
2. Adjustment knob
3. Indicator
4. CPO cyclic stick
5. Lever assembly
6. Lockout catch
7. Anchor ball
8. Pin
9. Fitting assembly

Figure 15. Cyclic Controls



1. TV/TIS select
2. Frame freeze
3. Nav/target designation
4. LOS control
5. Manual/slave
6. ICS/radio transmit
7. Point track
8. SCAS release
9. Trim release
10. Area track
11. Laser fire
12. FOV select

CPO Grip



13. Missile fire
14. MMS fixed forward
15. Missile select
16. Missile uncage
17. ICS/radio transmit
18. SCAS release
19. Trim release
20. Display select
21. Hover bob up
22. Spare

Pilot Grip

Figure 16. Pilot and CPO Cyclic Grips

Collective Control System

21. The collective control system is operated by a collective stick, located to the left of each crewmember seat and is the primary control for lift. A rotating grip-type throttle and a collective control head shown in figure 17 and photo 20 are located at the forward end of the pilot collective stick. The pilot's throttle has been desensitized by incorporating 171 degrees of rotation compared to 93 degrees of rotation for the OH-58A and OH-58C model helicopters. The copilot/observer (CPO) collective stick shown in figure 7 is removable. The storage location for the CPO collective stick is on the floor left of the CPO seat. Friction for the collective control system can be adjusted by rotating the friction adjustment knob located between the pilot and CPO seats.

Tail Rotor Control System

22. The tail rotor control system is operated by pilot or CPO anti-torque pedals. Pedal adjusters are provided to adjust pedal-to-seat distance for individual comfort.

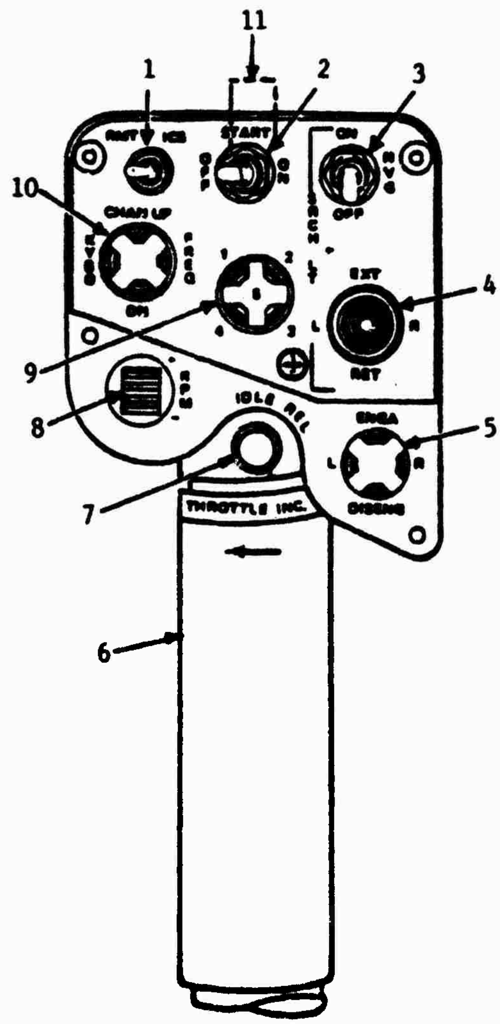
Force Trim System

23. A force trim system is incorporated in the cyclic and tail rotor control systems and is designed to provide artificial feel in the flight controls. The force trim system is activated by a FORCE TRIM switch located on the SCAS control panel as shown in figure 18. Force trim can be engaged or disengaged by the pilot. When heading hold (HDG HLD) is engaged, however, force trim is automatically engaged for tail rotor controls regardless of FORCE TRIM switch position. Force trim may be interrupted by pressing the TRIM REL button on either cyclic grip shown in figure 16 to allow repositioning of the cyclic or anti-torque controls to a new trim point.

STABILITY AND CONTROL AUGMENTATION SYSTEM

General

24. The SCAS is a three-axis (pitch, roll, and yaw) flight control augmentation system with a heading hold mode. The SCAS is a limited authority, rate reference system. The system is fail-safe in the pitch and roll axis (electrically shut down and actuators centered) and uses rate gyro, control motion, and airspeed inputs. The system is fail-operate (the malfunctioning system is electrically isolated) in the yaw axis for the first sensor or computer



1. Communication RMT ICS switch
2. START switch
3. Searchlight power switch
4. Searchlight control switch
5. SCAS heading hold ENGAge/DISENGage trim switch
6. Throttle
7. IDLE RELease switch
8. RPM trim switch
9. Radio select switch
10. CHANnel select switch
11. Start switch guard

Figure 17. Pilot's Collective Control Head

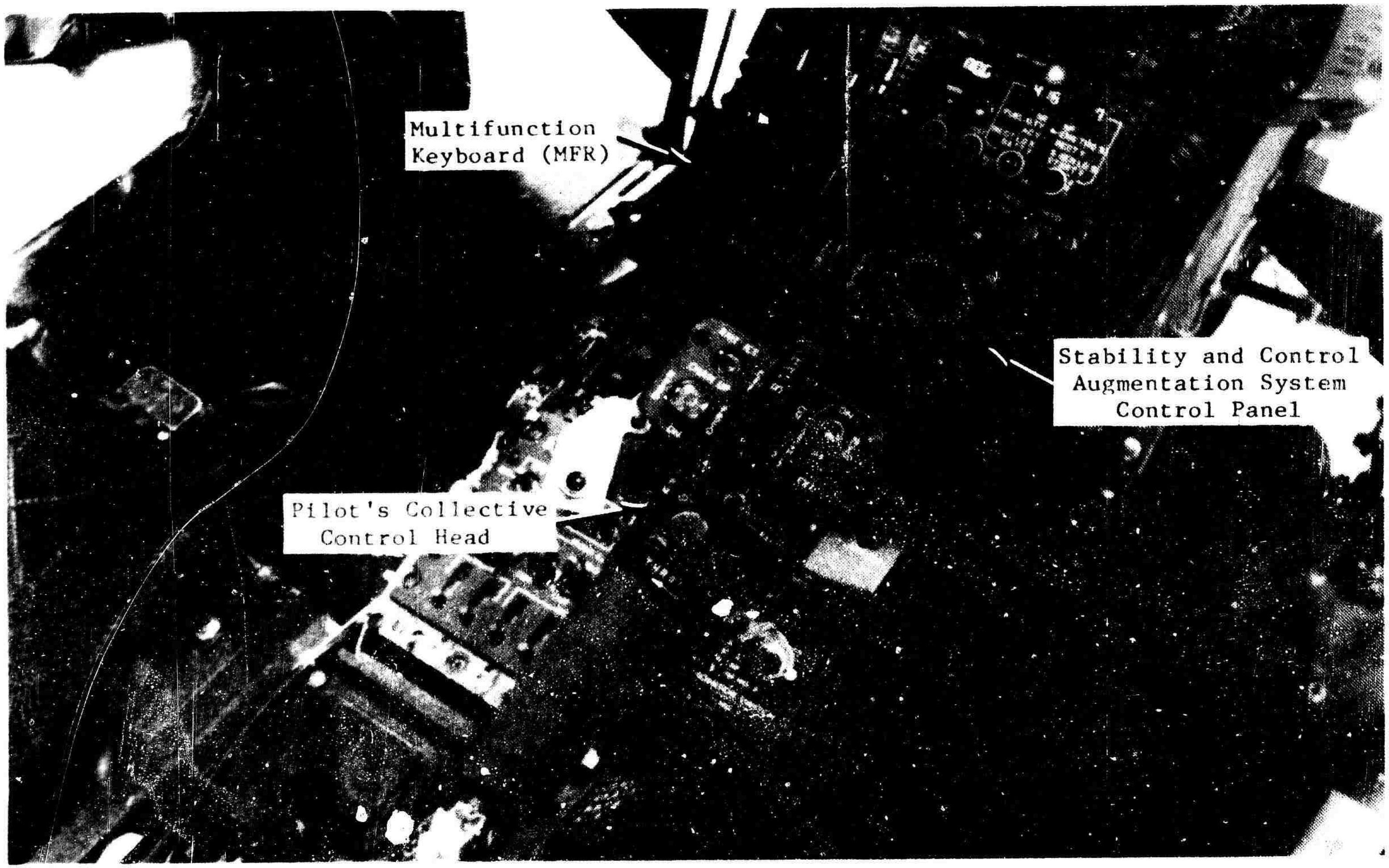


Figure 20. Center Pedestal

failure and fail-safe for subsequent yaw axis failures. During the conduct of the evaluation the roll SCAS gains were changed. Block diagrams indicating the gains in each SCAS channel, yaw SCAS/heading hold, and the mixer unit are shown in figures 19 through 23.

Control Panel

25. The SCAS control panel shown in figure 18 and photo 20 includes ENGAGE switches for pitch/roll and yaw channels, a PWR switch, a SCAS TEST switch, the FORCE TRIM switch, and the HYD SYST switch. The PITCH/ROLL and YAW ENGAGE switches apply power to the respective channels when the PWR switch is in the PWR (up) position. The TEST switch initiates the SCAS built-in-tests when the SCAS Check button has been selected on the multifunction display (MFD) and the engine is at idle.

SCAS Release Switch

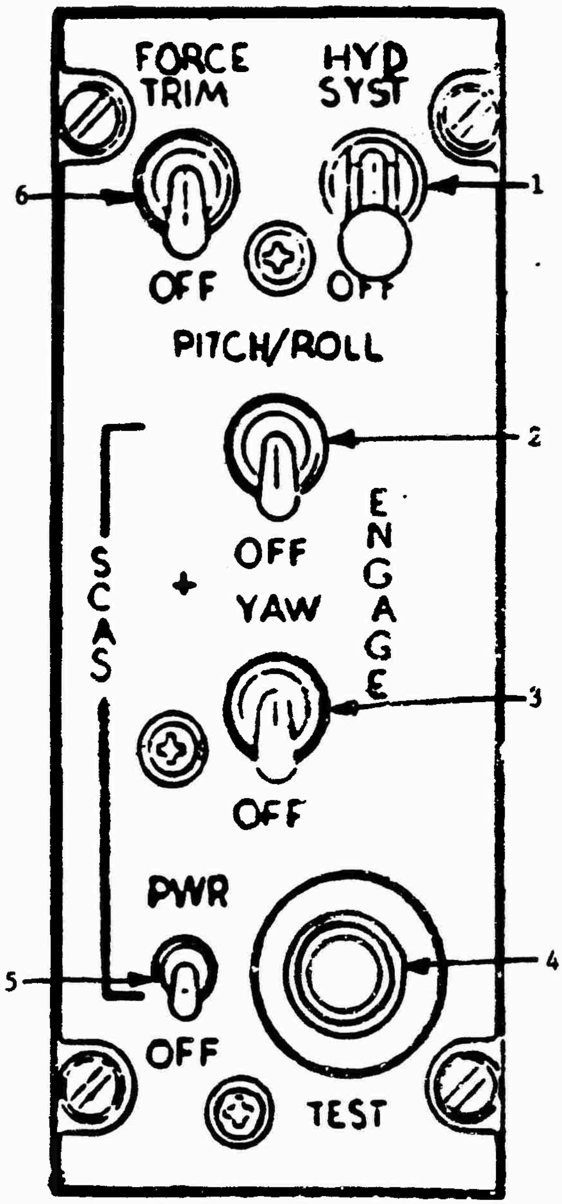
26. The SCAS REL switch is installed in each cyclic control grip, as shown in figure 16, designed to disengage pitch, roll and yaw with heading hold mode simultaneously. The system can be reactivated by reengaging the PITCH/ROLL and YAW ENGAGE switches.

SCAS Heading Hold Engage/Disengage Trim Switch

27. A SCAS heading hold ENGA DISENG trim switch is located on the pilot collective control head as shown in figure 17. The five-position, center spring switch is used to control the heading hold mode. The switch engages and disengages as well as references the aircraft heading left and right when heading hold is engaged.

Actuators

28. The SCAS uses two cyclic actuators and one yaw actuator. The two cyclic actuators are shown in photo 21 with the schematic shown in figure 24. The yaw actuator with SCAS is shown in photo 22 with the schematic shown in figure 25. The actuators are electro-hydraulic actuator assemblies which include a boost actuator and a SCAS actuator. The SCAS actuator includes a dual coil electro-hydraulic servovalve, a solenoid valve, centering springs, and locks. The solenoid valve is used to activate the SCAS actuator. If the solenoid valve is deenergized, the SCAS actuator is designed to center and lock.



1. Hydraulic system switch
2. Pitch/roll engage switch
3. Yaw SCAS engage switch
4. Auto preflight test button
5. Power switch
6. Force trim switch

Figure 18. Stability and Control Augmentation (SCAS) Control Panel

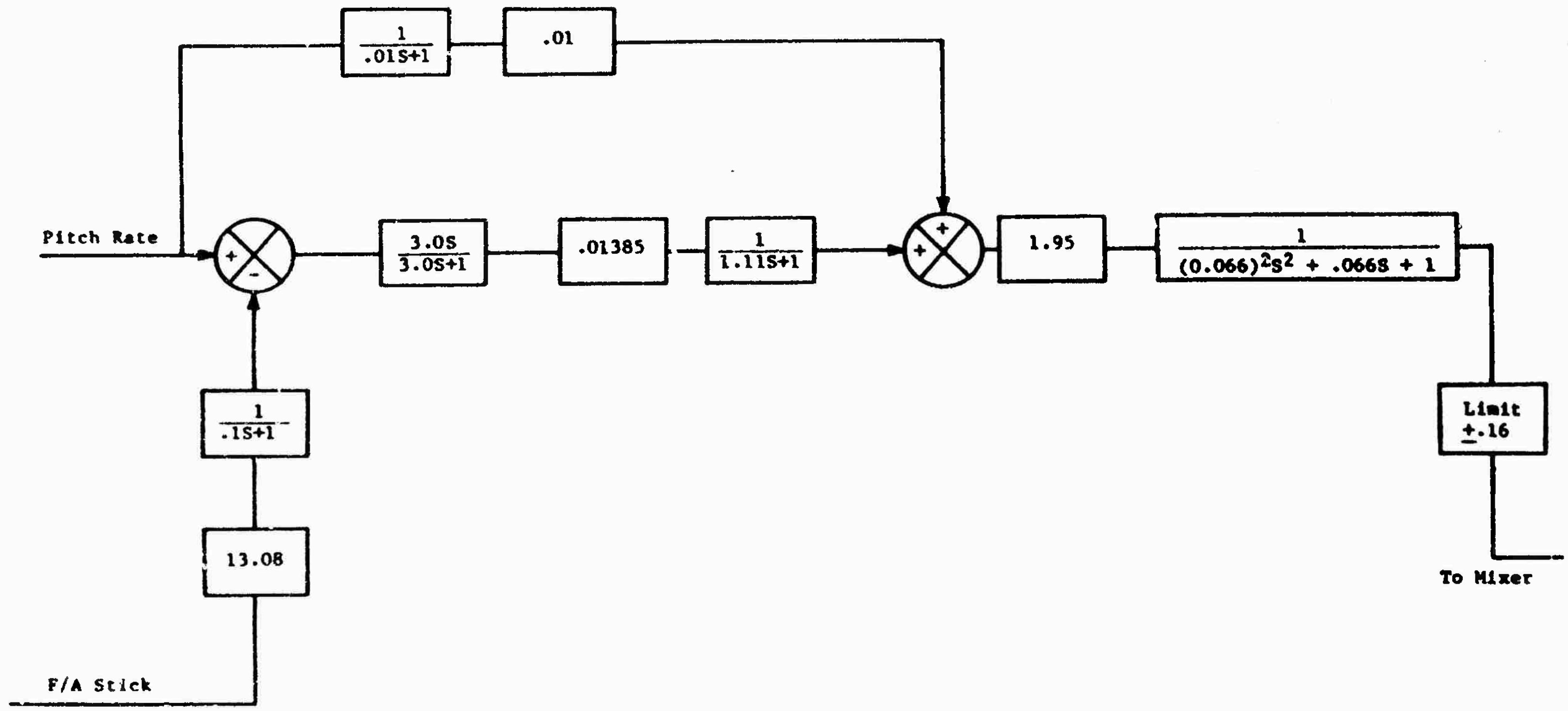
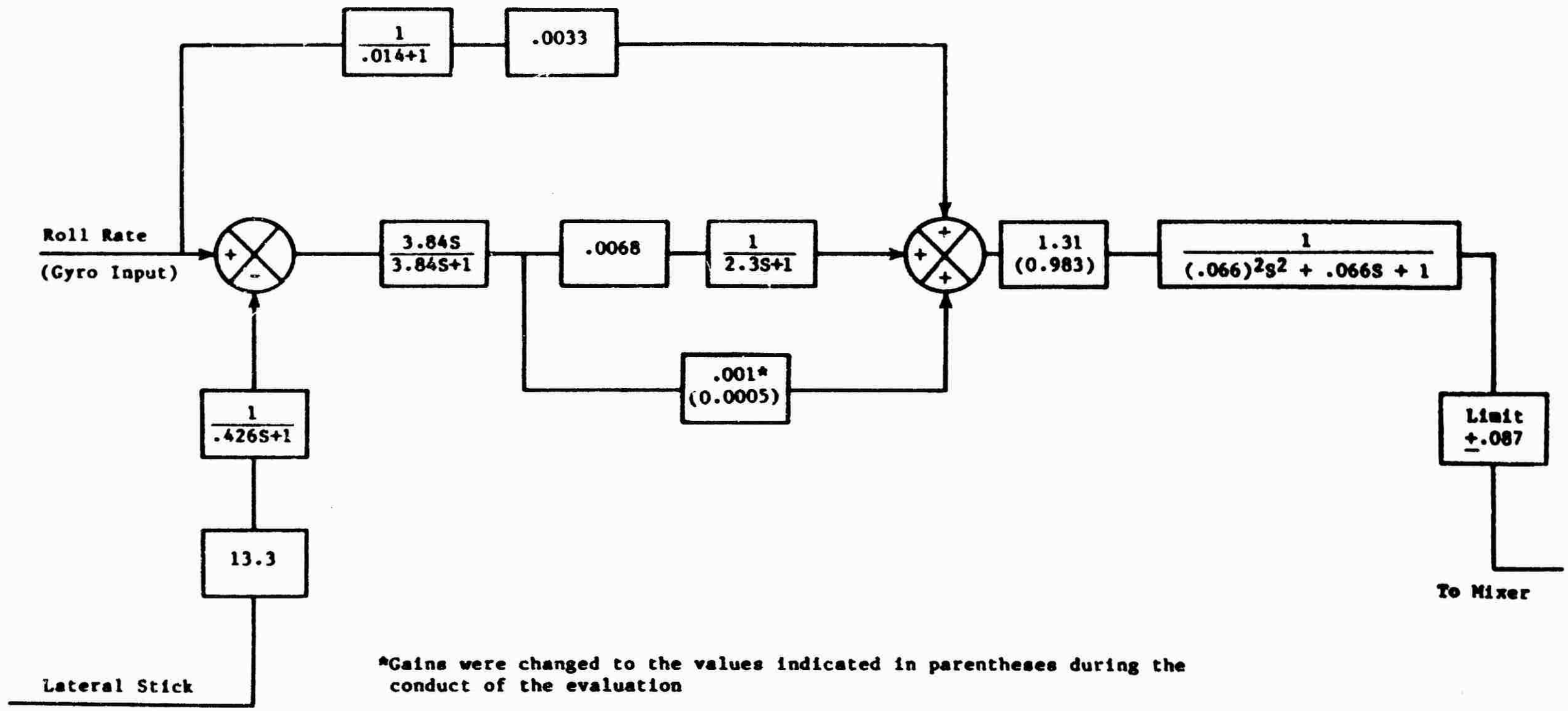


Figure 19. Pitch SCAS Diagram



*Gains were changed to the values indicated in parentheses during the conduct of the evaluation

Figure 20. Roll SCAS Diagram

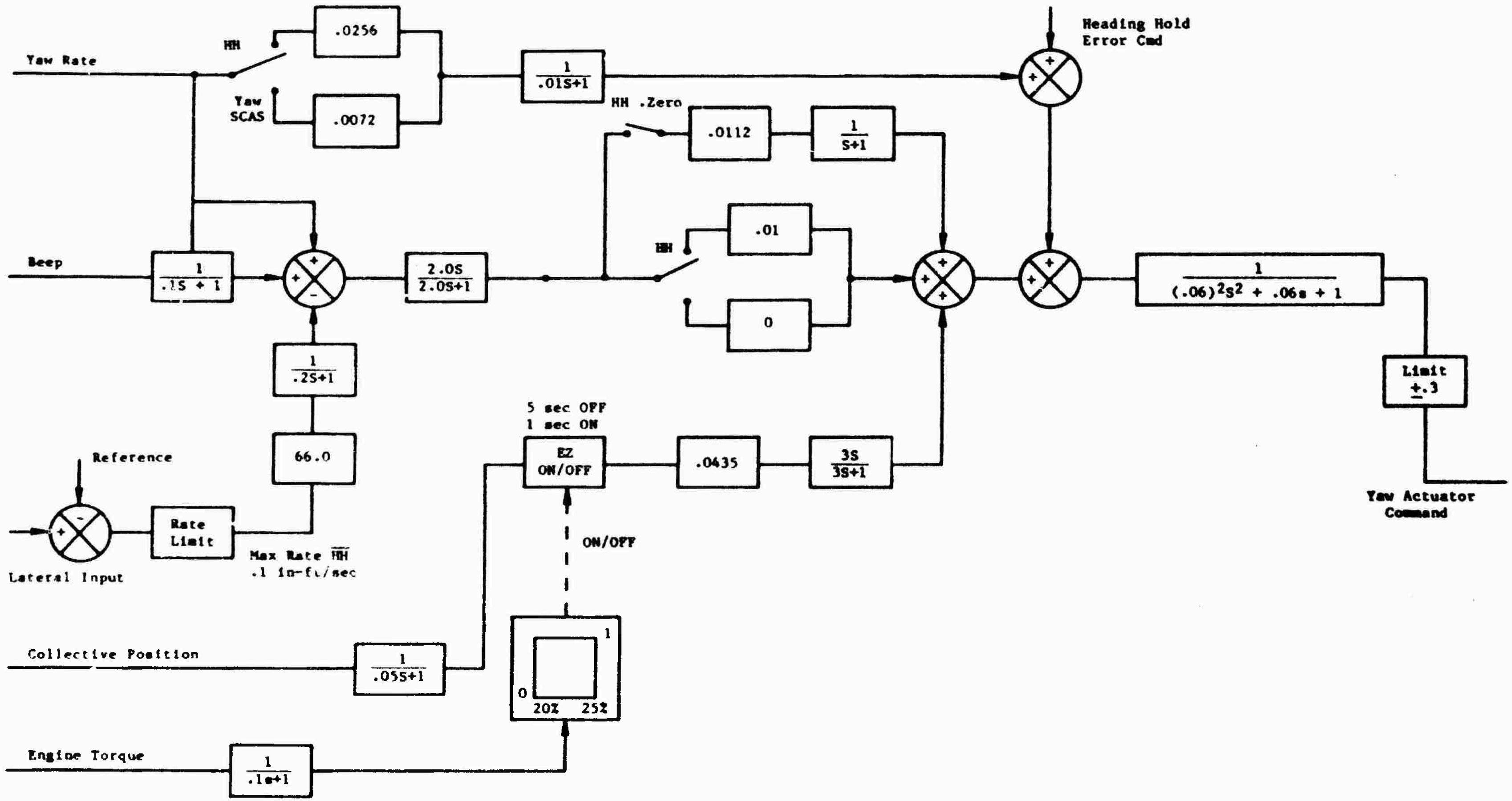


Figure 21. Yaw SCAS Diagram

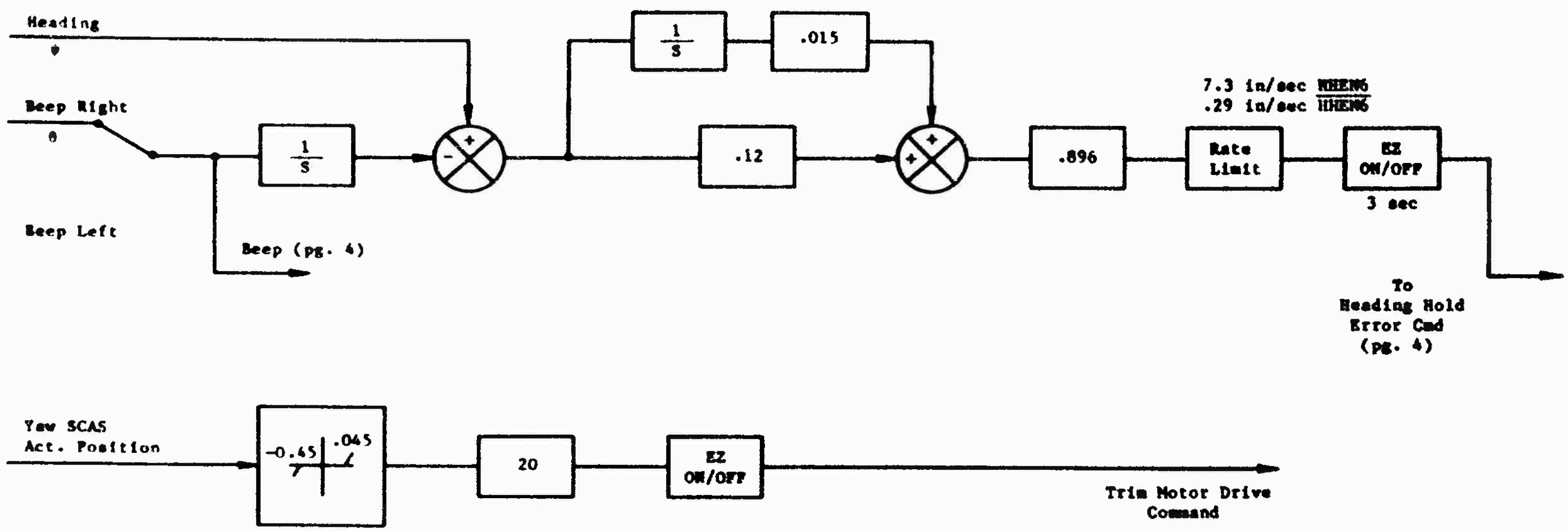


Figure 22. Yaw SCAS/Heading Hold Diagram

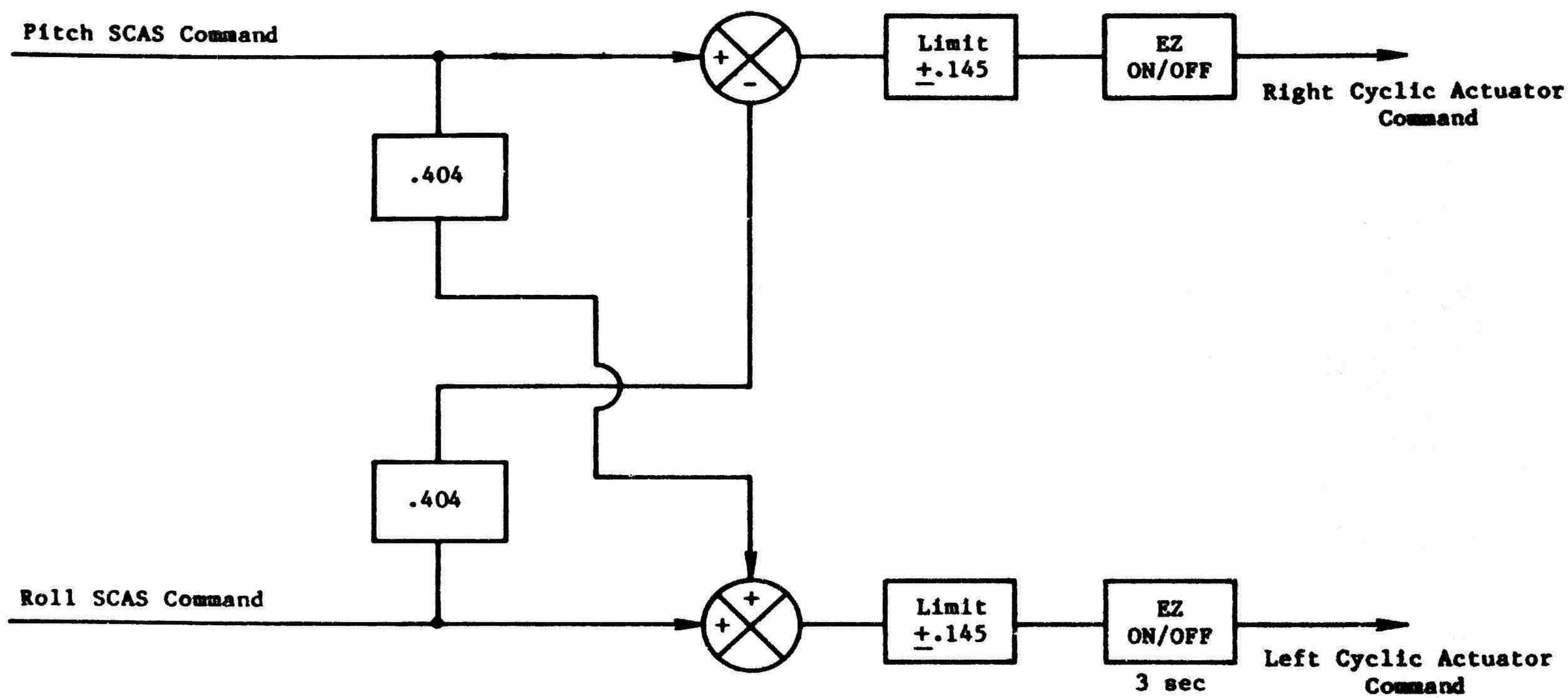


Figure 23. Mixer Diagram

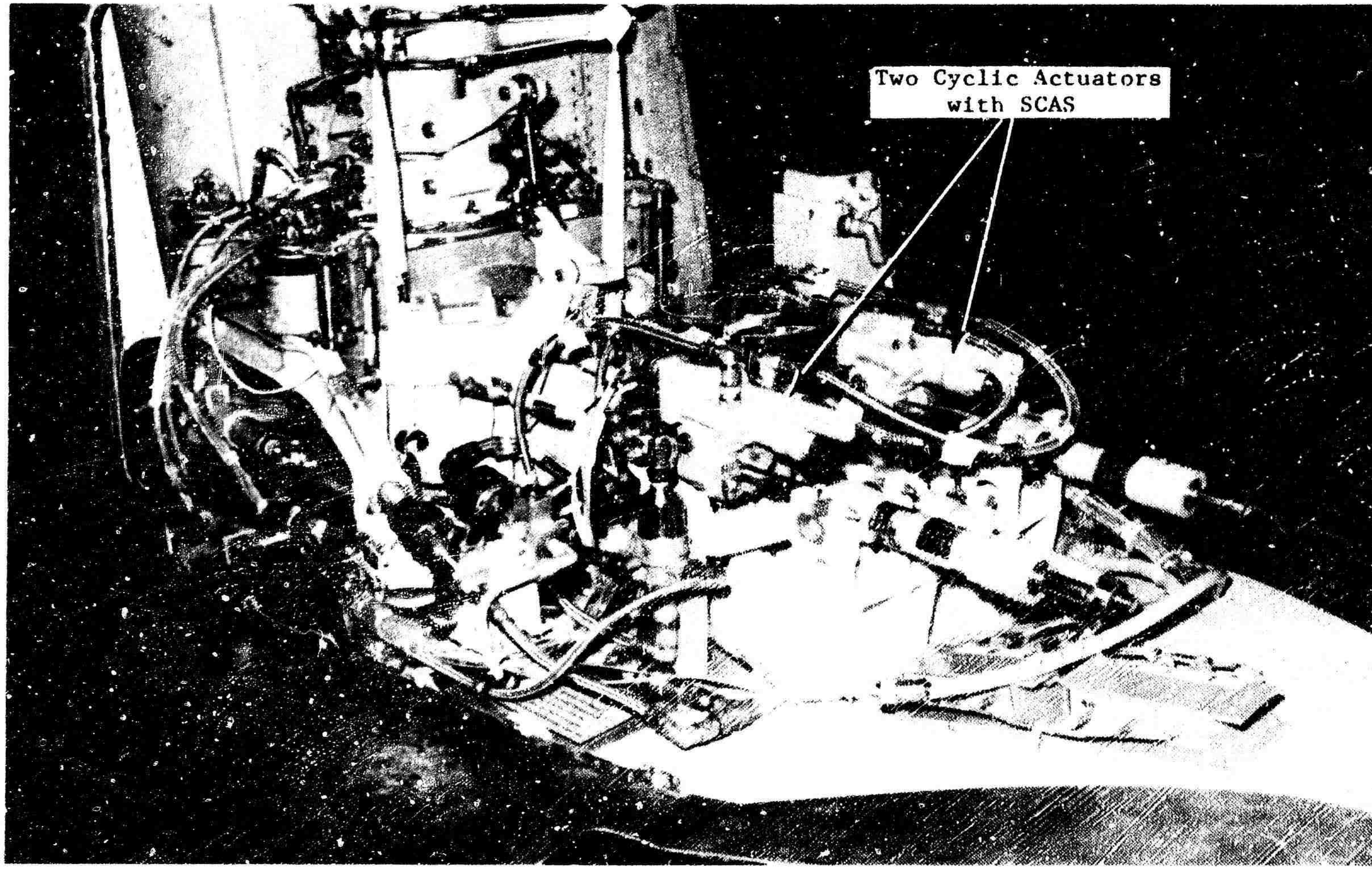


Photo 21. Cyclic Actuators with SCAS

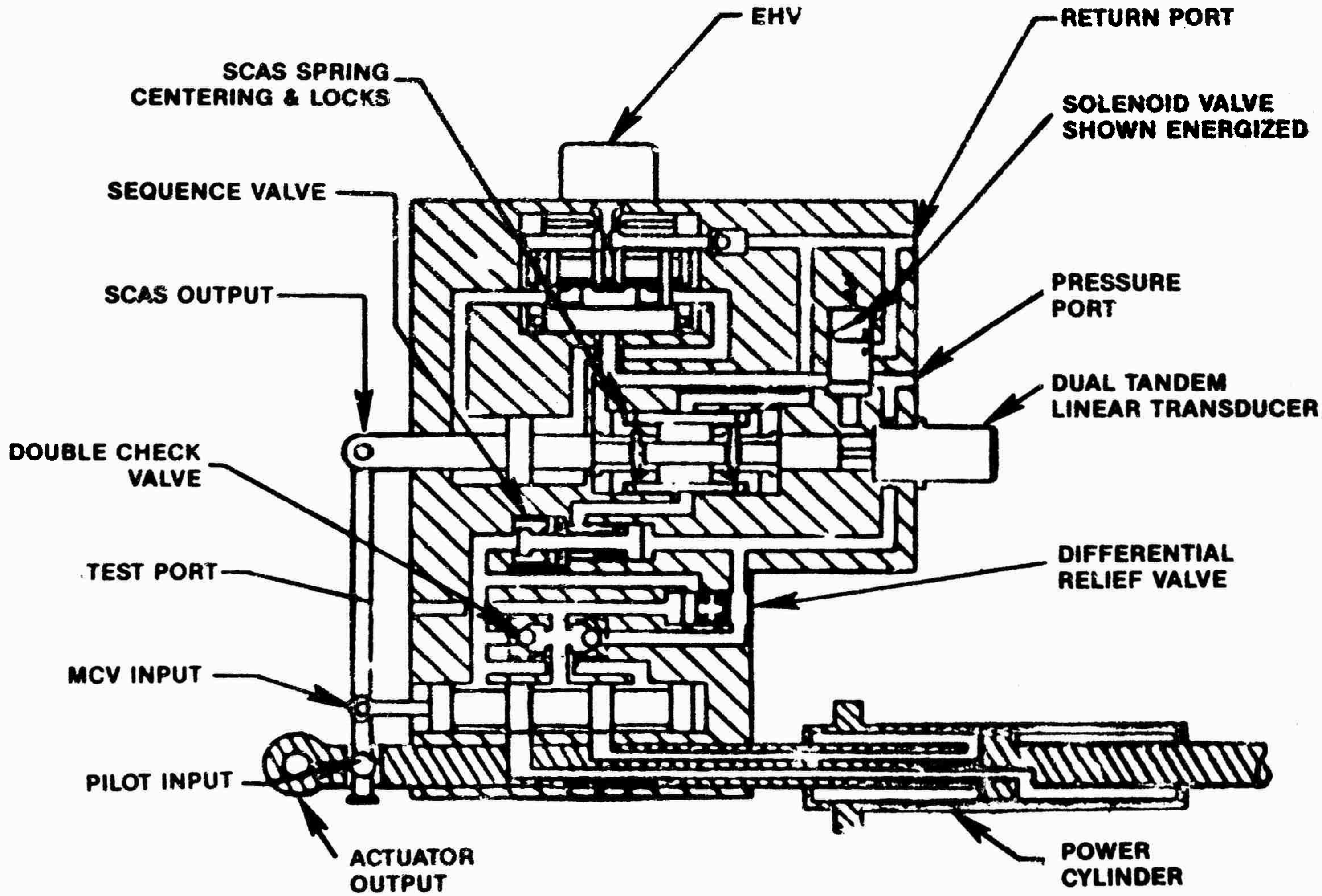


Figure 24. Cyclic Actuator Schematic

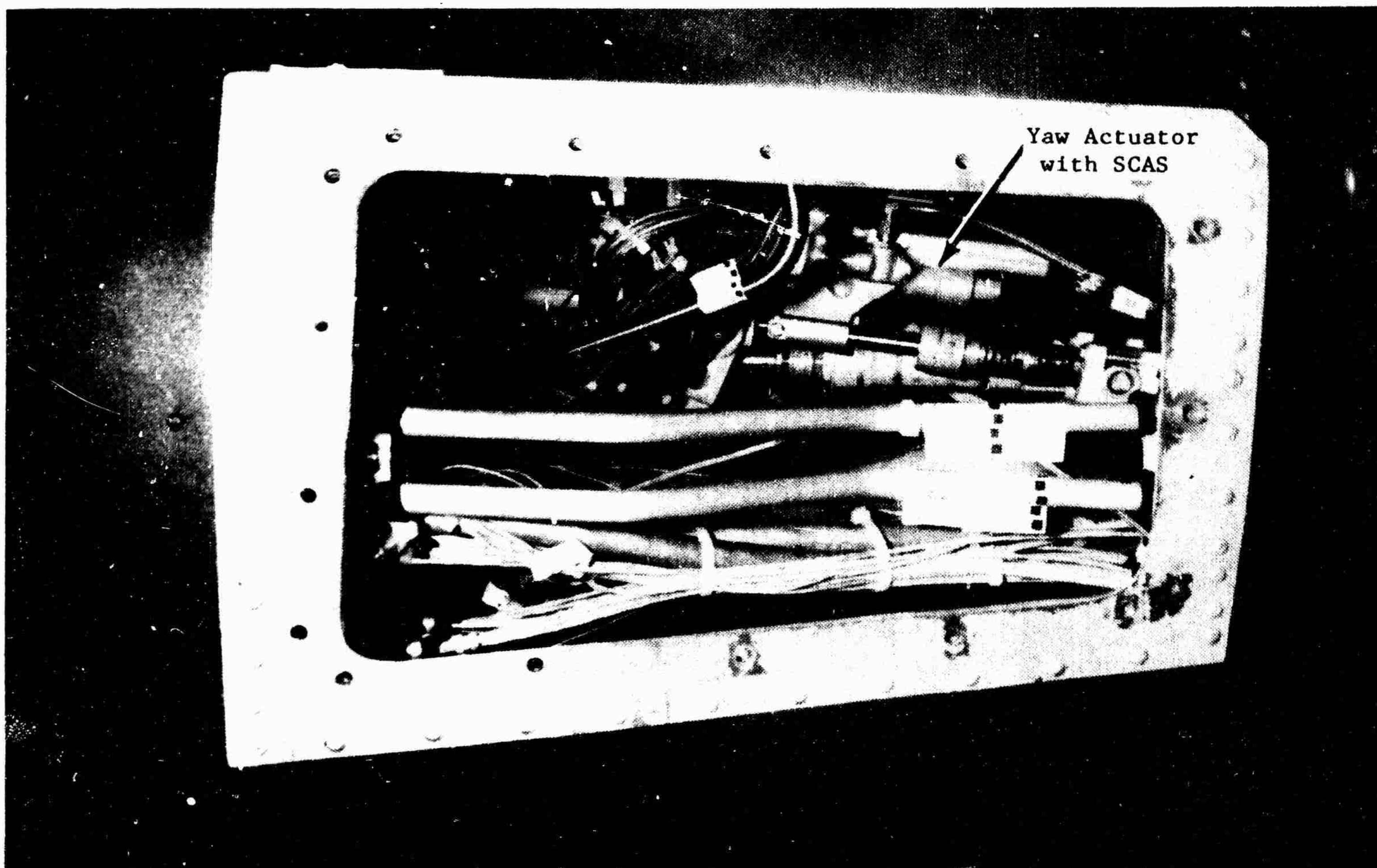


Photo 22. Yaw Actuator with SCAS

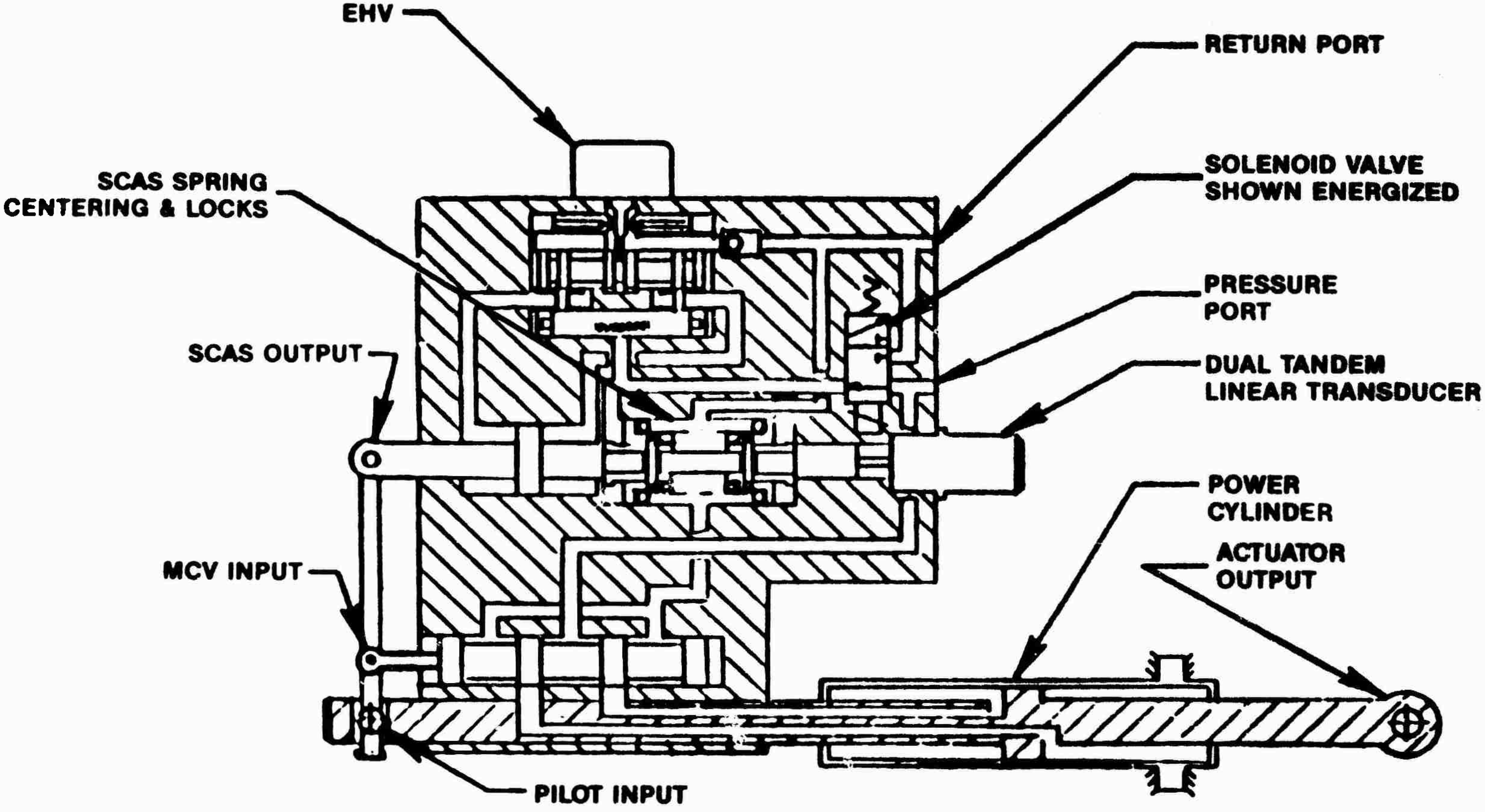


Figure 25. Yaw Actuator Schematic

SCAS Actuator Authorities

29. The SCAS actuator strokes are limited to give the following SCAS authorities of full control travel.

Pitch	$\pm$6.0% (electronic limits) $\pm$6.7% (mechanical limits)
Roll	$\pm$6.0% (electronic limits) $\pm$11.0% (mechanical limits)
Yaw	$\pm$10.0%

The full control authority (0 to 100%) at the respective rotor is as follows:

Pitch	21.5 degrees
Roll	10.8 degrees
Yaw	36.1 degrees

FUEL SYSTEM

General

30. The fuel system shown in figure 26 consists of a crash resistant, self-sealing fuel cell, one engine driven fuel pump, one fuel cell mounted priming/boost pump, quantity indication system, fuel control unit, digital control unit (DCU), fuel filter, fuel injector, emergency shutoff valve, drainage provisions, low level warning system, filter bypass indicator, and breakaway valve. Total fuel capacity is 109.8 U.S. gallons with 104 gallons being usable.

Fuel Boost Pump

31. A fuel boost pump, located in the fuel cell as shown in figure 26, is provided for assisting the engine driven suction pump to deliver fuel and is automatically engaged during engine starting. The fuel boost pump shall be on when operating above 10,000 ft pressure altitude.

Fuel Low Caution Message

32. A FUEL LOW caution message is designed to illuminate on the MFD when there is approximately 97.5 pounds (15 gallons) of fuel remaining.

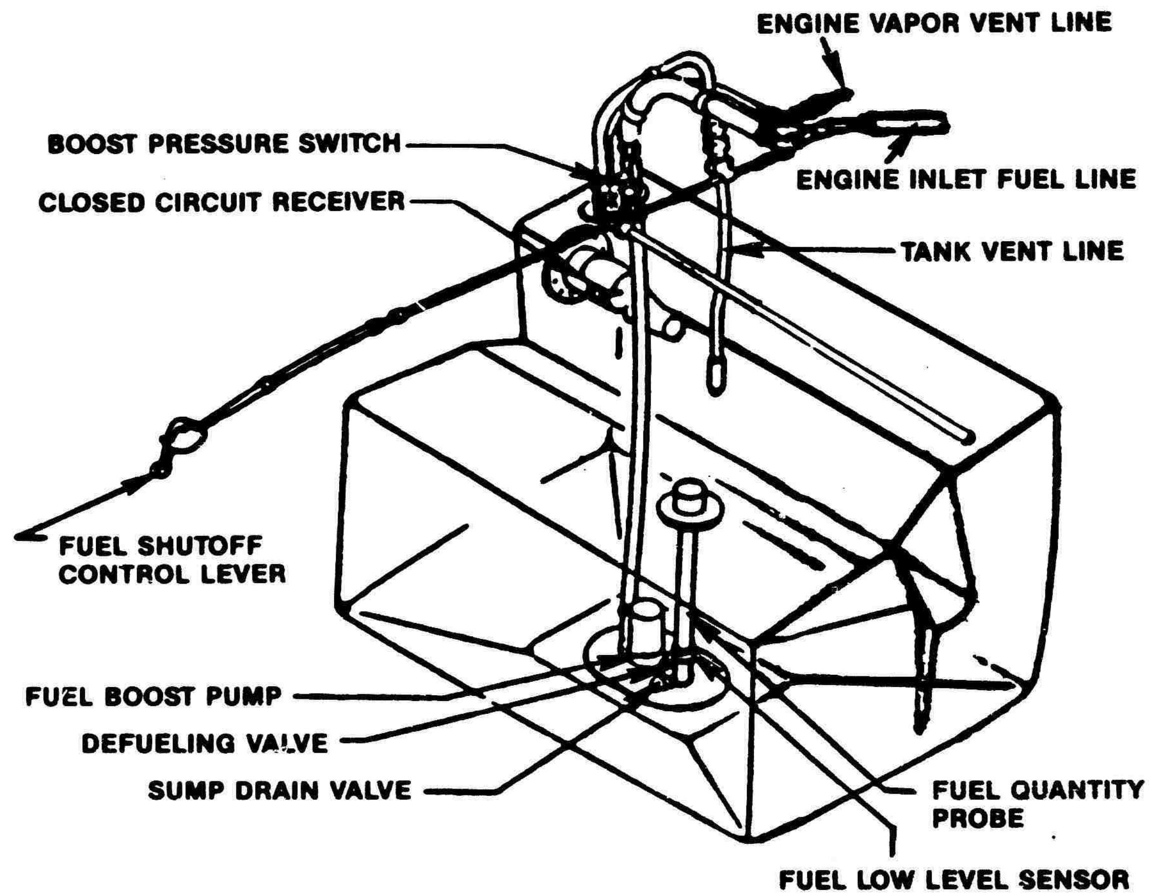


Figure 26. Fuel System

Fuel Control Panel

33. The fuel control panel shown in figure 27 is located on the right side of the center pedestal. It contains a covered NORM-ANLG BACK UP switch labeled DIGITAL and ANALOG. The NORM-ANLG BACK UP switch is used to manually select the digital control unit or analog governor. The push type DIGITAL test switch is used to test the overspeed control of the digital control unit. The ANALOG test switch is designed to check the operation of the overspeed control of the analog system.

Digital Control Unit

34. The digital control unit electronically meters fuel for the engine fuel system. The DCU is powered by the battery-emergency bus.

35. The DCU is designed to minimize transient droop during power applications and control rpm surges and turbine overspeeds.

36. The backup mode of DCU operation is the analog mode and designed to automatically assumes responsibility for engine fuel management in the event of digital mode failure. In the analog mode, the pilot cannot control the power turbine speed with the rpm trim switch. RPM surge protection is also lost resulting in the possibility of an engine overspeed during rapid rotor unloading conditions.

37. In the event of a total electrical failure or failure of both the digital and analog governors, the electronic controls are deenergized allowing manual control of fuel flow. In this mode the engine operates on the gas producer speed (Ng) governor in the fuel control unit and the pilot controls the engine power output by manual throttle control.

MULTIFUNCTION DISPLAYS

38. The pilot and CPO MFD located as shown in figure 28 are interchangeable and operated in the same manner. Each MFD has an 8-inch diagonal cathode ray tube (CRT), five software keys down each side and four software keys across the bottom of the bezel as shown in figure 29. A fault indicator is located at the top center of the bezel to alert the crew of a MFD failure. Should one of the systems fail, the brightness knob can be pulled out to the backup position to monitor what is on the other MFD. The MFD's normally operate independently allowing separate functions to be performed by the crewmembers. The MFD's are used to

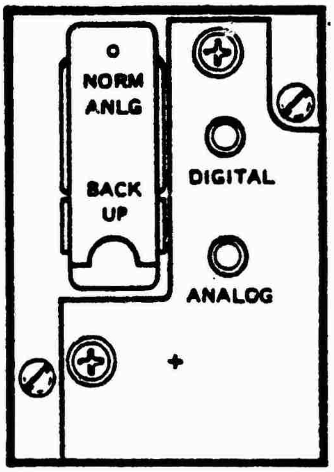


Figure 27. Fuel Control Panel

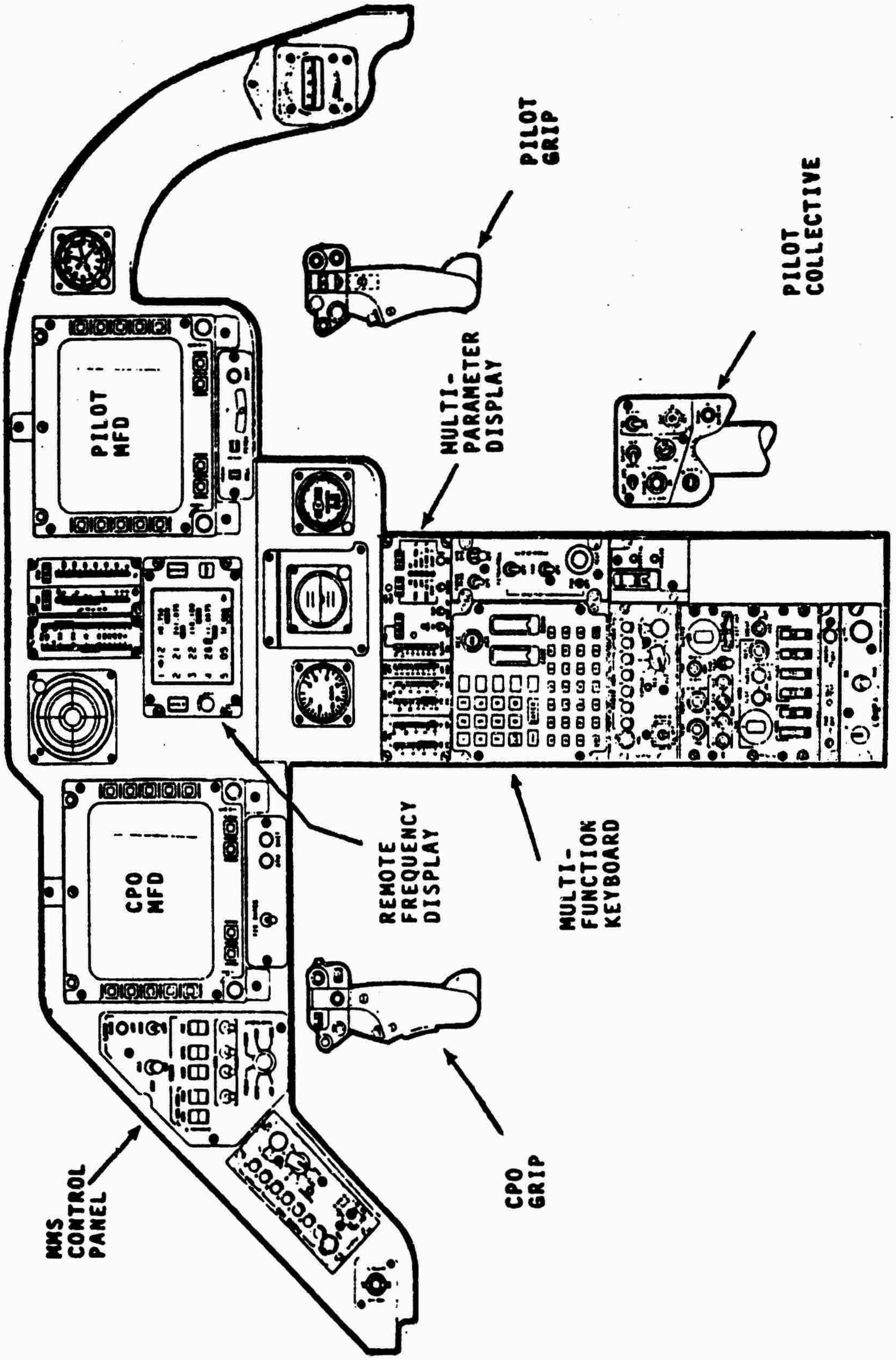


Figure 28. Crewstation

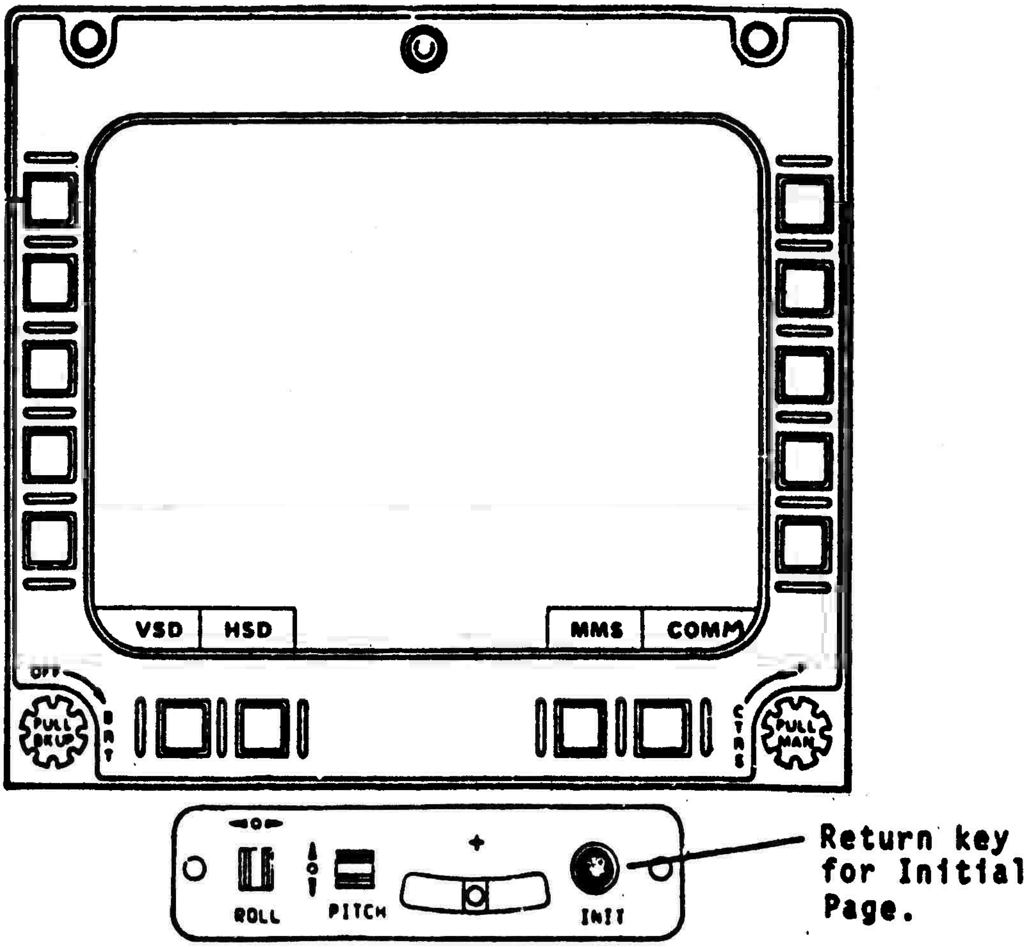


Figure 29. Multifunction Display (MFD)

display navigation, communication, flight sensor and caution/warning information as well as video for mast mounted sight operations.

MULTIFUNCTION KEYBOARD

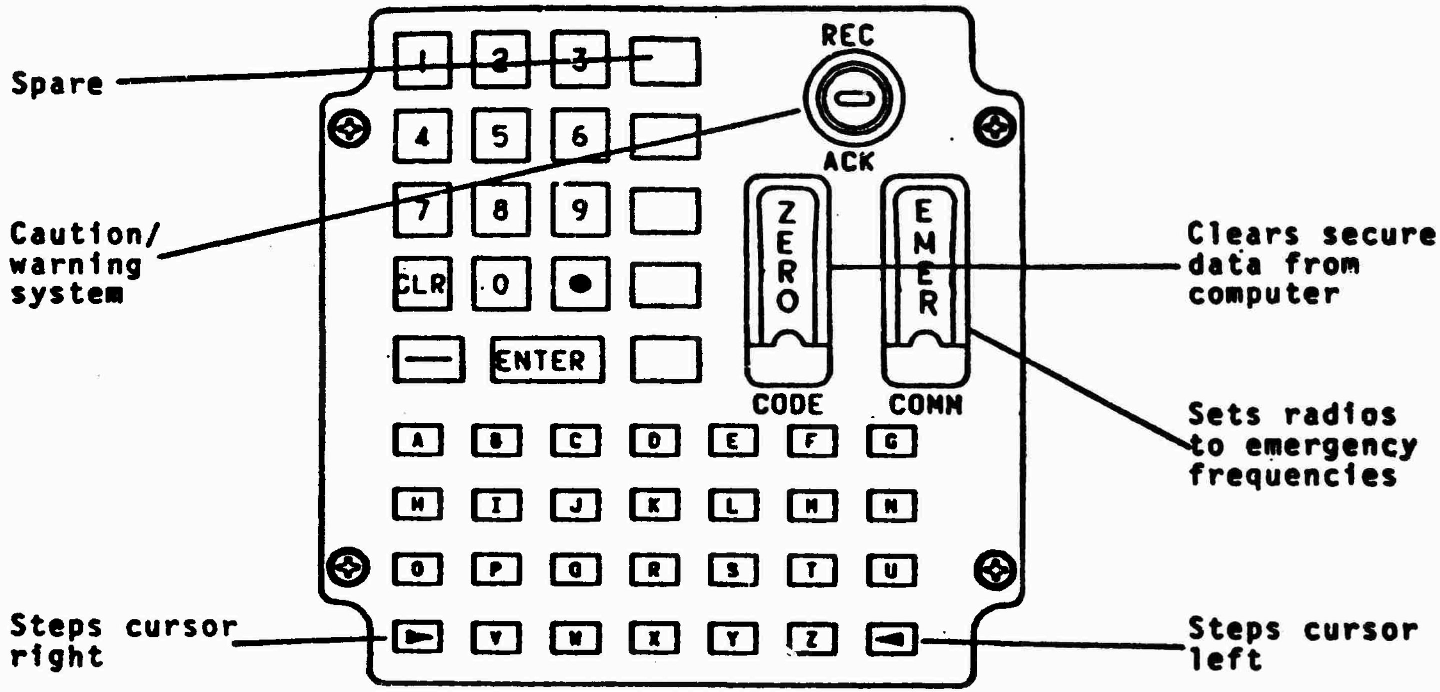
39. The multifunction keyboard (MFK) shown in figure 30 and photo 20 is located on the lower console immediately below the multiparameter display (MPD) panel as depicted in figure 27, and is accessible by both crewmembers. The keyboard is tied into both MFD's as a source of entering data such as communication frequencies and channels, waypoint indentifiers, coordinates, navigation data and test mode codes. The two guarded switches provide a means of clearing sensitive data in emergencies and selecting emergency radio frequencies rapidly. The ACK/REC switch is associated with the caution/warning system.

REMOTE FREQUENCY DISPLAY

40. The remote frequency display (RFD) shown in figure 31 and photo 23 in the center of the instrument panel between the two MFD's as shown in figure 28. The display screen is a liquid crystal panel designed to be readable under all lighting conditions and with night vision compatability. The RFD serves as a frequency display for the five communication radios. The screen displays the present channel number, frequency, cipher mode, and identifies which radio each crewmember is tuned for transmission. The RFD is tied into the MFD's and the MFK through the master controller processor unit (MCPU). A TEST switch located on the lower right portion of the RFD, has two functions: changing input from one MCPU to another and providing evidence of proper RFD display.

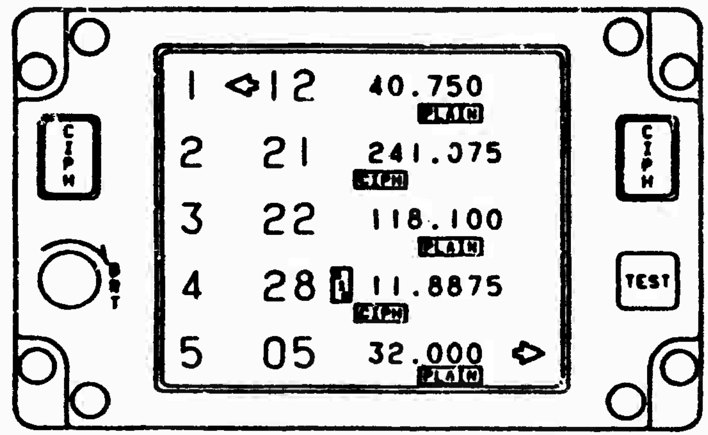
SYSTEM INSTRUMENTS

41. The Vertical Scale Display Subsystem (VSDS) consists of the following: the MPD, dual tachometer and turbine gas temperature (TGT)/mast torque (TRQ) indicator. The displays are vertical scale type indicators, either alone or in combination with digital readouts. The instruments for system monitoring are solid state, light-emitting vertical scale instruments. Each vertical scale indicator contains a multisegment illuminated vertical scale readout against a fixed graduated scale. The bottom segment in all of the indicators is a power-on indicator lamp and is located below the scale graduations. All vertical scale instruments



The keyboard is accessible to both crewmembers and is used to access the computer for entering data. Navigation waypoints, radio channels and frequencies and other data are entered through the keyboard. The acknowledge/recall switch is associated with the caution/warning system. The two guarded switches provided a means of clearing sensitive data in emergencies and selecting emergency radio frequencies rapidly. The unlabeled keys are spares.

Figure 30. Multifunction Keyboard (MFK)



The Remote Frequency Display (RFD) is a liquid crystal display and is located in the center of the instrument panel. It constantly displays the status of each of the aircraft's five radios.

Radio channel selection is normally made by the communications control switch located on the pilot's collective head and on the left side of the observer's instrument panel.

Radio selection may also be made through the pilot's or observer's CSC panel.

Channel and frequency selection may also be made through the individual communications control pages.

Figure 31. Remote Frequency Display (RFD)

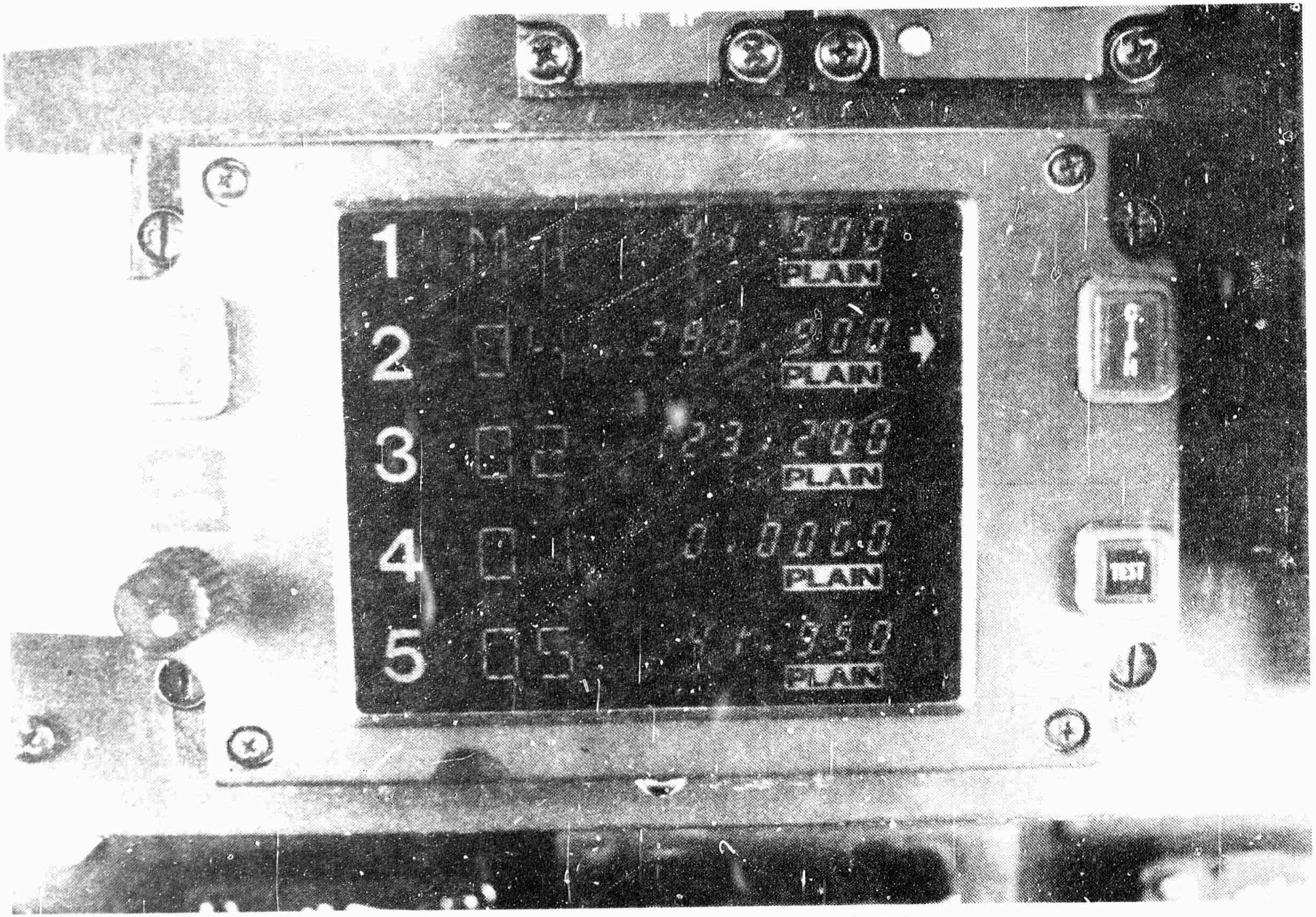


Photo 23. Remote Frequency Display (RFD)

have range markings to show operating limits, caution limits, and avoidance areas. A built-in-test (BIT) switch allows performance of BIT on the VSDS. A TEST switch allows a test to be performed on the digital displays, light segments, and lamps and also allows the crew to delete the digital displays. The vertical scale instruments have light sensors for automatic adjustment of display brightness for compatability with cockpit ambient light. One sensor is located on the lower end of the TGT indicator and the other sensor is located below the digital readout window for the Ng. The two sensor values are summed and the display brightness is automatically adjusted for the average of the two.

42. The MPD, shown in figure 32 and photo 24, is located on the pedestal as shown in figure 28 and contains vertical scale instruments containing 16-segment for transmission oil pressure and temperature, engine oil pressure and temperature, fuel quantity, and NG. The NG speed is also read out on a digital display. The right side of the MPD provides selectable digital readouts for parameters that are not displayed elsewhere. The selectable parameter display allows the selection of any of the following five pairs of digital readouts as desired by pressing the SEL switch.

a. Rotor speed (N_R) and power turbine speed (N_P).

b. Fuel quantity (FUEL QTY) and engine torque (ENG TRQ %).

c. AC voltage (AC V) and rectifier voltage (RECT V).

d. Rectifier load (RECT LD %) and starter-generator load (S GEN LD %).

e. Battery voltage (BATT V) and starter voltage (START V).

43. The dual tachometer indicator, shown in figure 33, is a vertical scale instrument containing a 33-segment display located on the upper center portion of the instrument panel as shown in photo 25. The left scale indicates main rotor speed (N_R) in percent. Rotor speed signals are received from the mast torque signal processor through the MPD. The right scale indicates power turbine speed (N_P) in percent.

44. The TGT/TRQ indicators shown in figure 23, include a digital display in conjunction with a 28-segment vertical scale display and are located on the upper center portion of the instrument panel as shown in photo 25. The left scale indicated TGT in degrees Celsius (C X 100). TGT is sensed at the gas producer turbine outlet. The right scale indicates main rotor mast torque

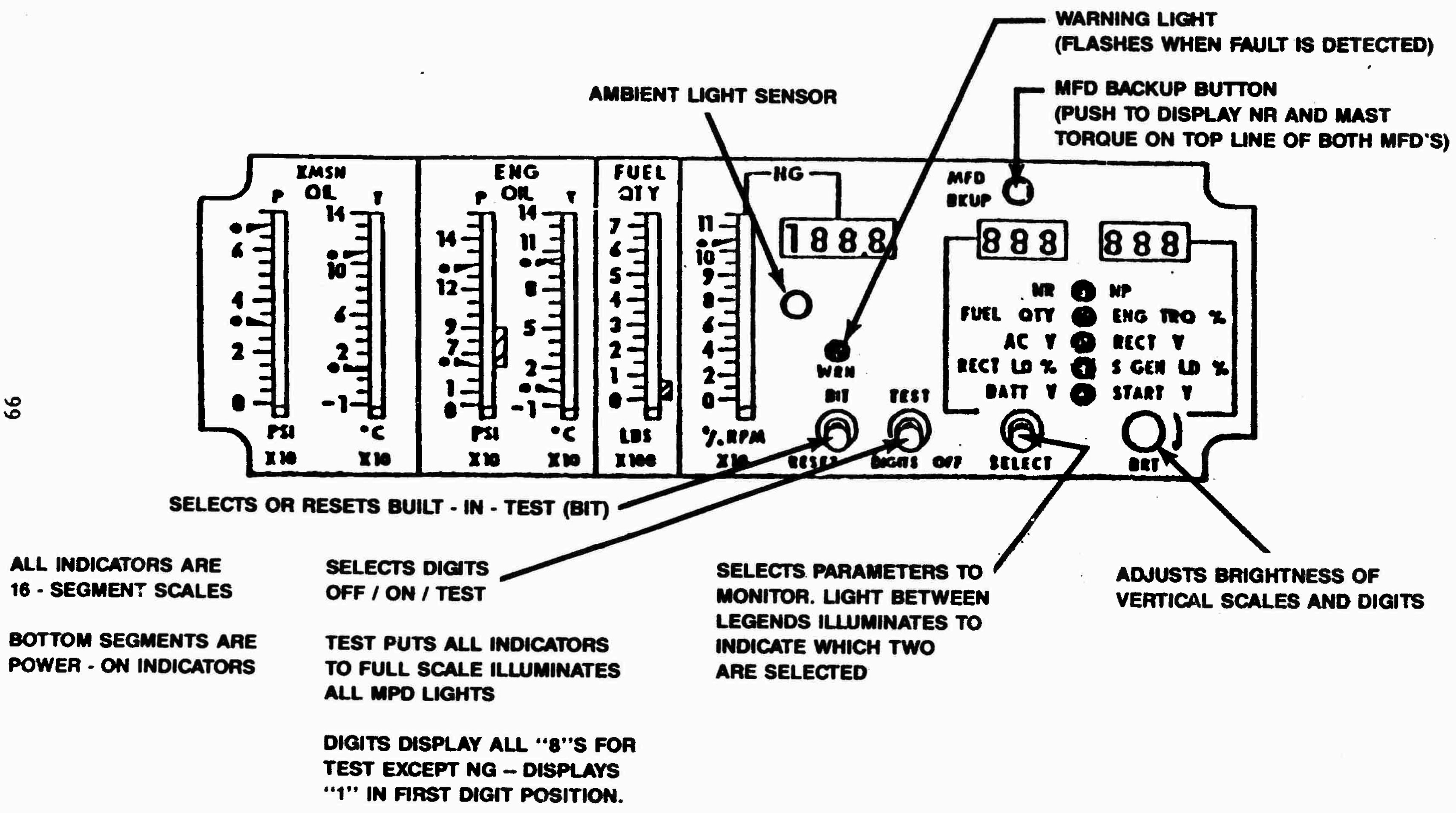


Figure 32. Multiparameter Display (MPD)

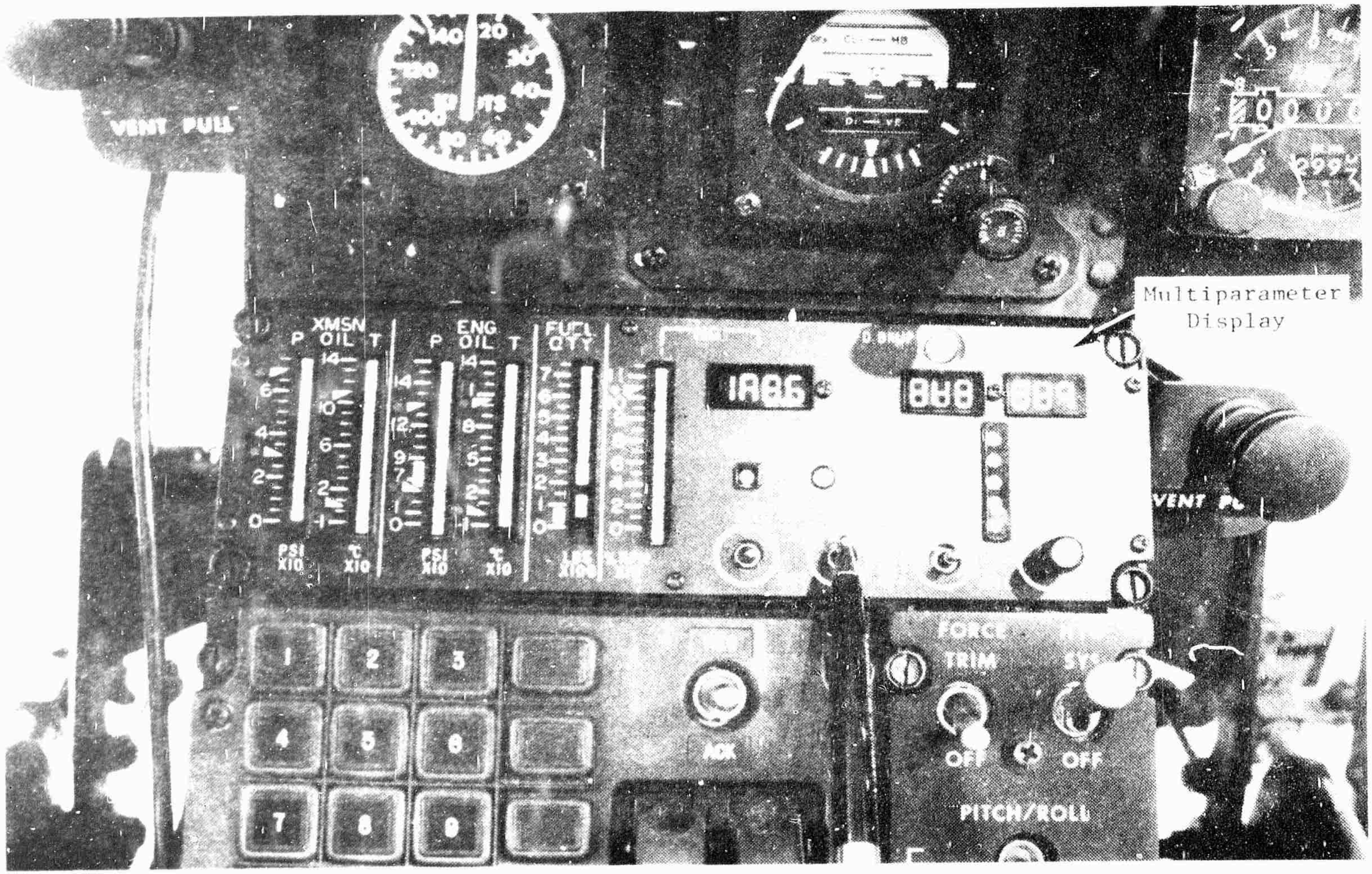
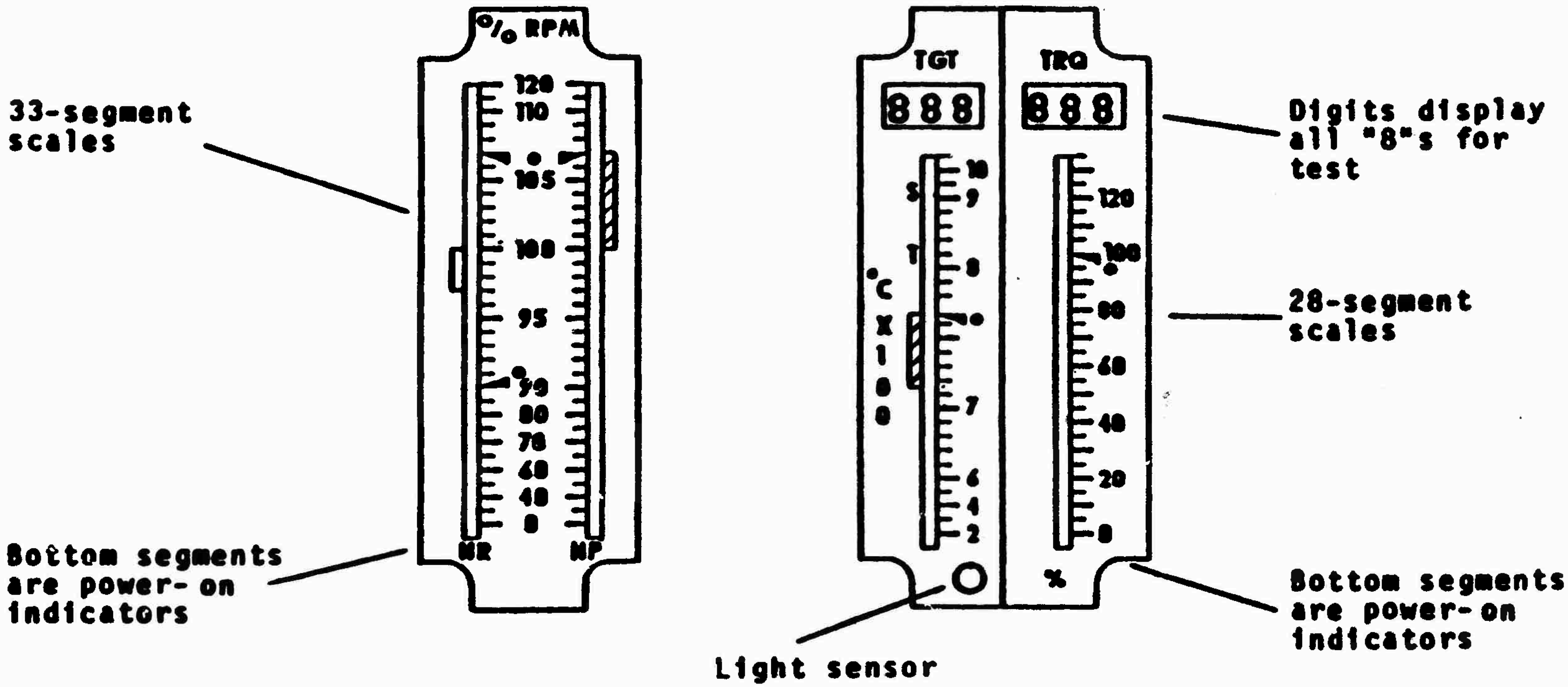


Photo 24. Multiparameter Display (MPD)



Digital Ranges

TGT range 000 - 999

Torque range 000 - 140

Figure 33. Dual Tachometer and TGT/Torque Indicators

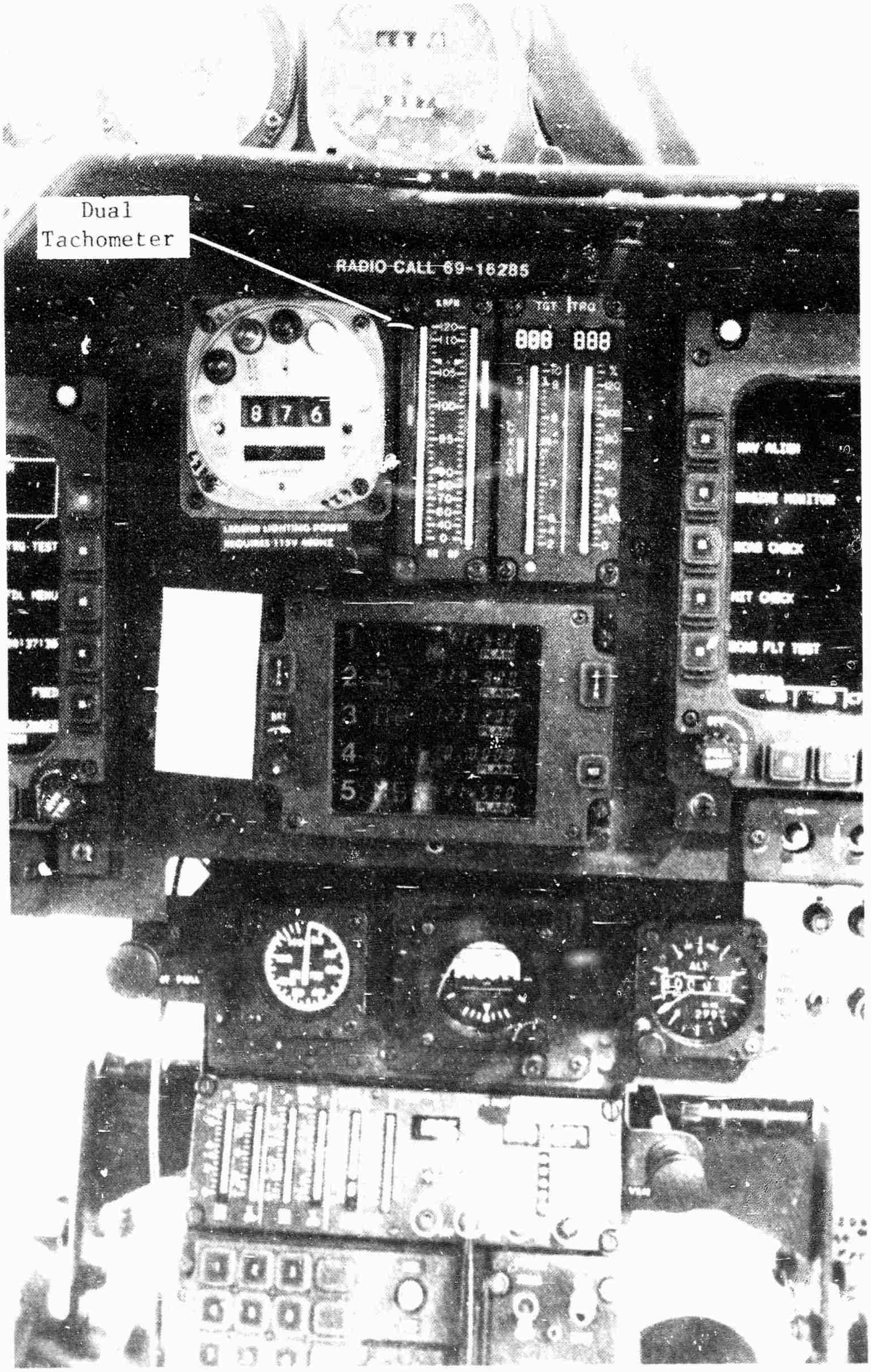


Photo 25. Dual Tachometer and TGT/Torque Indicator

in percent. Torque indication is received from one upper and two lower mast torque sensors through the MPD shown in figure 34.

45. In the event of a failure of the MPD, rotor speed and mast torque can be displayed across the top of both MFD screens by pressing the MFD BKUP switch on the MPD.

HYDRAULIC SYSTEM

46. The hydraulic system is designed to reduce crewmember effort and fatigue by minimizing cyclic, collective, and anti-torque control input and feedback forces. The test helicopter incorporated an auxiliary hydraulic system performing as a backup hydraulic system. The complete hydraulic system is shown in the schematic in figure 35.

ELECTRICAL SYSTEM

47. Primary AC electrical power for the helicopter systems is provided by a 120/208 volt, three-phase, 400 Hz, air cooled AC generator. This generator is driven by the engine power turbine from an engine accessory drive pad.

48. Primary DC electrical power for the helicopter systems is provided by two sources. The DC essential bus is powered from a 28 volt, 200 ampere transformer rectifier unit (TRU) and the battery emergency bus is powered by a 28 volt, 200 ampere starter-generator.

49. Backup systems for both DC and AC primary power systems are provided. The backup systems are designed to ensure that no single electrical failure will cause the loss of any system essential to tactical instrument flight. Automatic switching from primary power to backup is provided. Backup AC power is provided by the starter-generator. In the event the AC generator fails, the starter-generator input to an inverter is designed to supply AC power. A starter-generator failure results in the TRU assuming the full load imposed by the DC essential bus, battery emergency bus and power assured bus. In the event of an AC generator failure, the inverter is designed to assume the loads on the 115V AC essential bus.

50. Electrical power for starting and emergency power supply is provided by a single 24 volt, 17 ampere battery located in the nose of the helicopter. Complete provisions are included for the installation of a second battery. When installed, the second battery is located in the aft electrical compartment.

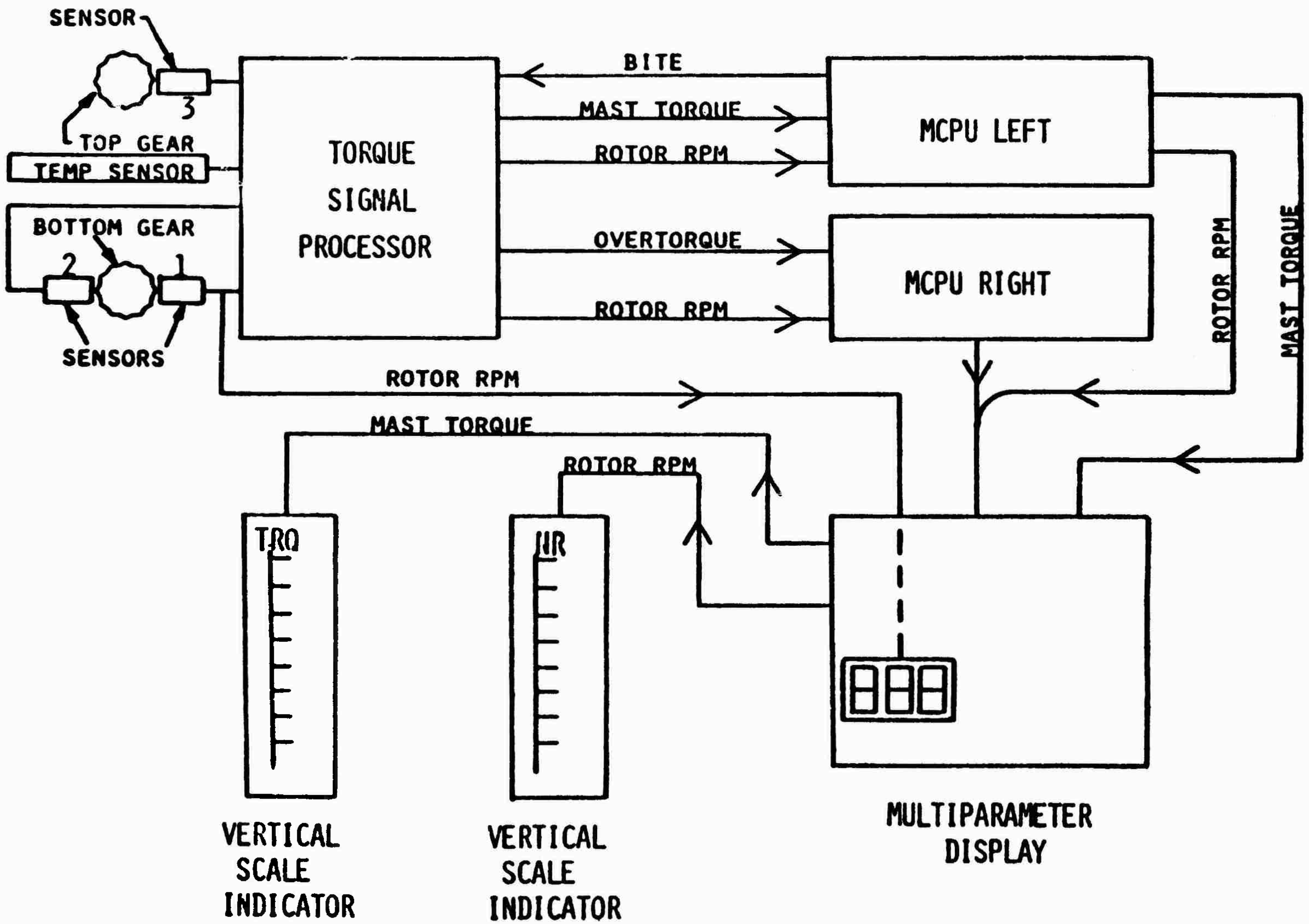


Figure 34. Mast Torque System Block Diagram

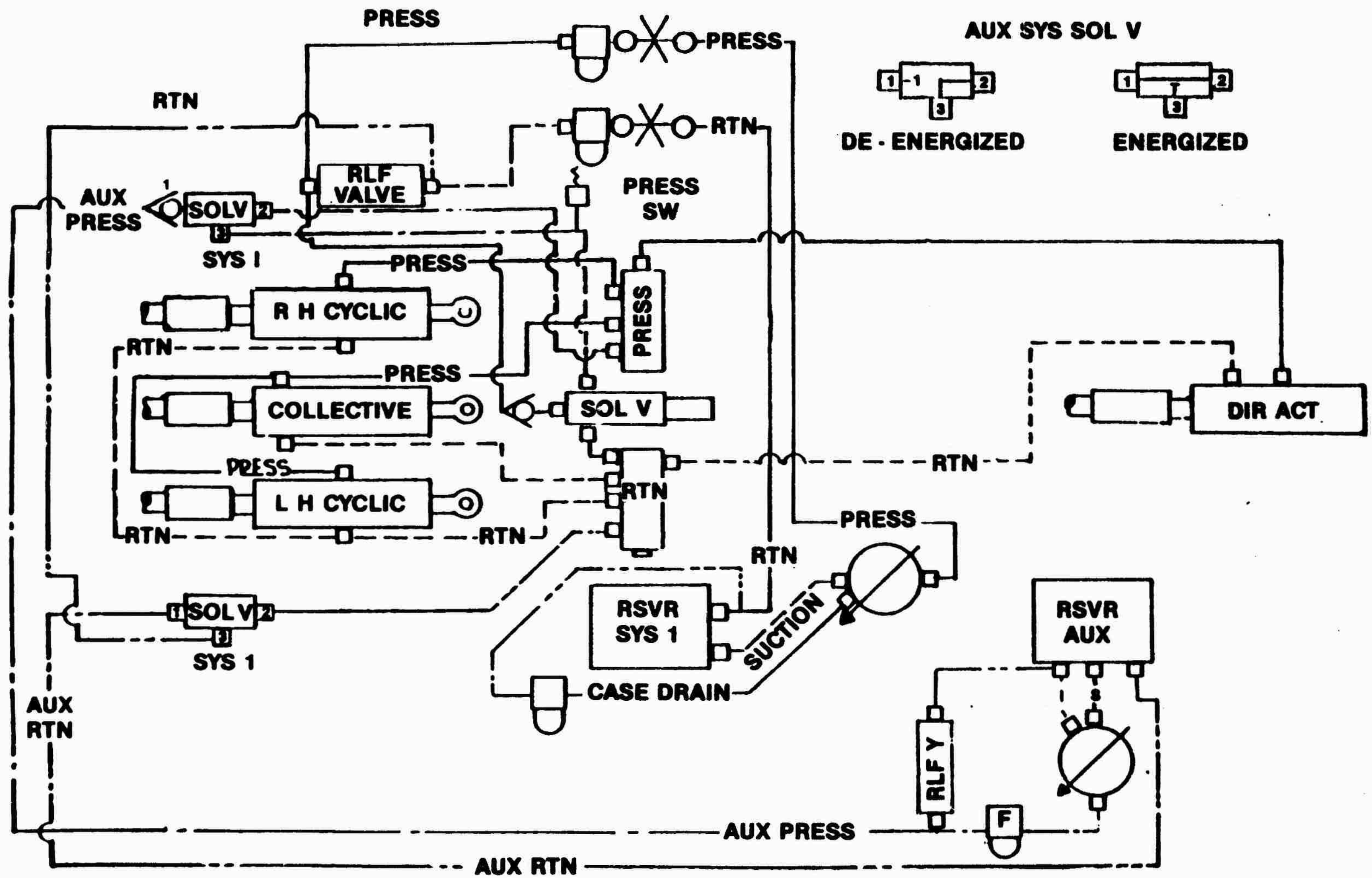


Figure 35. AHIP Hydraulic System Schematic with Auxiliary Hydraulic System

51. External power receptacles are provided for application of both AC and DC electrical power while on the ground. With DC external power applied, engine starting and systems check out may be accomplished. Application of AC external power allows operation and check out of systems powered by 115 VAC and 26 VAC power.

CONTROL SYSTEMS RIGGING CHECK

52. A rigging check of the cyclic, collective, and directional controls was performed in accordance with BHTI drawing number 406-401-001 as modified by Limited Design Variation number 406EA2269. The rigging measurements were within the tolerances of these documents. Additionally, measurements were made of the mast angle relative to the aircraft fuselage, and of the swashplate relative to the mast at the neutral cyclic rig point. The following measurements were read:

Mast angle relative to fuselage
- lateral - 1 degree, 25 minutes left tilt (in a fuselage station plane)
- longitudinal - 4 degrees, 56 minutes forward tilt (in a fuselage butt line plane)

Swashplate angle relative to mast at neutral cyclic rig
- lateral - 1 degree, 5 minutes left tilt
- longitudinal - 1 degree, 4 minutes forward tilt

WEIGHT AND BALANCE

53. Prior to testing the aircraft, gross weight and longitudinal and lateral center of gravity (cg) were determined using calibrated scales. The aircraft was weighed with trapped fuel, full oil, no ballast, no crew, in the primary mission configuration with instrumentation installed and the multi-purpose lightweight missile system removed. The results of this weighing were:

Gross weight - 3335 pounds
Longitudinal cg - FS 117.65
Lateral cg - BL 0.32 right

54. The fuel loading for each test flight was determined prior to engine start and following engine shutdown using a test fuel quantity gage. Fuel used in flight was recorded by a test fuel-used system and verified with the pre- and postflight determinations.

55. Aircraft gross weight and cg were controlled by ballast installed at various locations in the aircraft.

APPENDIX C. INSTRUMENTATION

1. Test instrumentation was installed, calibrated, and maintained by Bell Helicopter Textron, Inc. Data was displayed in the cockpit, and recorded on magnetic tape onboard the aircraft.

2. A tubular, aluminum boom was mounted on the aircraft and extended forward from the nose. The boom held a swiveling pitot-static tube and sensors for angles of attack and sideslip. The boom airspeed calibration is presented as figure A in this appendix.

3. Parameters measured during this test were:

Control positions
- Longitudinal
- Lateral
- Directional
- Collective
- Engine condition selector

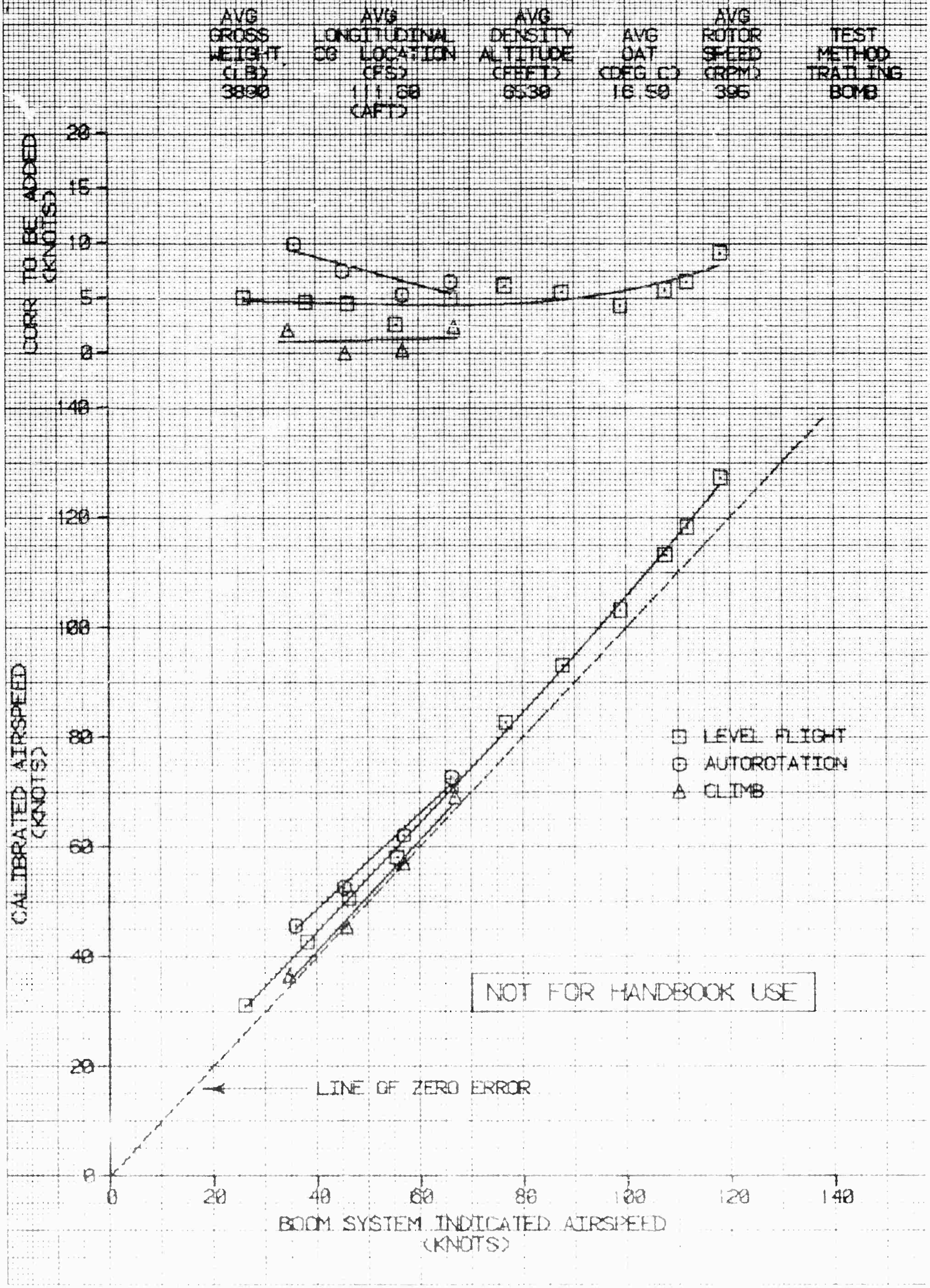
Stability and control augmentation system actuator positions
- Pitch
- Roll
- Yaw

Aircraft attitudes, angular velocities, and angular accelerations
- Pitch
- Roll
- Yaw

Airspeed (boom and standard ship systems)
Altitude (boom and standard ship systems)
Altitude (radar height above ground level)
Free air temperature
Angle of attack
Angle of sideslip
Main rotor speed
Engine gas generator speed
Engine turbine outlet temperature
Engine torque
Main rotor torque
Tail rotor torque
Fuel flow rate
Fuel used
Center of gravity normal acceleration
Vibration accelerations
- Pilot seat (FS 70, WL 21, BL 17R)
  - Longitudinal
  - Lateral
  - Vertical
- Copilot seat (FS 70, WL 21, BL 17L)
  - Longitudinal

Lateral
Vertical
Event marker (pilot position)
Tape record number
Time code

FIGURE A
BOOM SYSTEM AIRSPEED CALIBRATION
OH-58D USA S/N 69-16235



APPENDIX D. TEST TECHNIQUES AND DATA ANALYSIS METHODS

AIRSPEED CALIBRATION

1. The test boom and standard ship's pitot-static systems were calibrated using the trailing bomb method. The airspeed measured by the calibrated trailing bomb suspended below the test aircraft was used as reference for determining position error. The boom airspeed calibration is presented in figure A, appendix C.

HANDLING QUALITIES

Control Positions in Trimmed Forward Flight

2. Control positions and aircraft pitch attitude as functions of airspeed were determined during level flight performance. For airspeeds above the maximum level flight airspeed, engine power was set to maintain the rotor mast torque limit.

Static Longitudinal Stability

3. The static longitudinal stability tests were accomplished by establishing the trim condition in ball-centered flight and then varying control positions to obtain airspeed changes about the trim airspeed with collective control held fixed at the trim value. The airspeed range of interest was approximately +20 knots from trim. Altitude was allowed to vary as required during the test.

Static Lateral-Directional Stability

4. These tests were conducted by establishing the trim condition and then varying sideslip angle incrementally up to the preestablished limits. During each test, collective control position and airspeed were held constant and altitude allowed to vary as required.

Maneuvering Stability

5. This test was accomplished by establishing the trim condition and then incrementally increasing load factor by increasing roll attitude while holding airspeed and collective control position constant and allowing altitude to vary as necessary. Turns were made in both directions. Pullup and pushover maneuvers were also used to evaluate the maneuvering stability.

Dynamic Stability

6. Dynamic longitudinal and lateral-directional stability were qualitatively evaluated to determine both the short- and long-period characteristics. The short-period response was evaluated by use of longitudinal, lateral, and directional pulse or doublet inputs and by releases from steady-heading sideslips. The long-period dynamic response was evaluated longitudinally by slowly returning the flight controls to trim position following a change of 10 knots indicated airspeed from the trim airspeed and then holding controls fixed while recording the aircraft response.

Controllability

7. Controllability testing was conducted by first establishing a trim condition and then making a step-type control input which was held until the aircraft had reached a steady rate. Inputs of varying size were made in each direction of the longitudinal and lateral cyclic controls and the directional control.

8. Data were analyzed by first reading from time history data of the maneuver the following parameters:

- Control input size
- Maximum angular velocity achieved
- Time from initial angular velocity changes to 63% of the maximum angular velocity
- Attitude change in one second from control input (1/2 second for roll)
- Maximum angular acceleration
- Time from control input to maximum angular acceleration

9. Angular velocity damping was calculated taking the reciprocal of the time to 63% maximum angular velocity.

VIBRATION

10. The output of vibration accelerometers at selected locations in the aircraft were recorded on magnetic tape onboard the aircraft. These data were then reduced using a fast Fourier transform method to obtain average vibration amplitudes as a function of frequency. Amplitude at main rotor harmonic frequencies were then determined and plotted.

DEFINITIONS

Qualitative Rating Scales

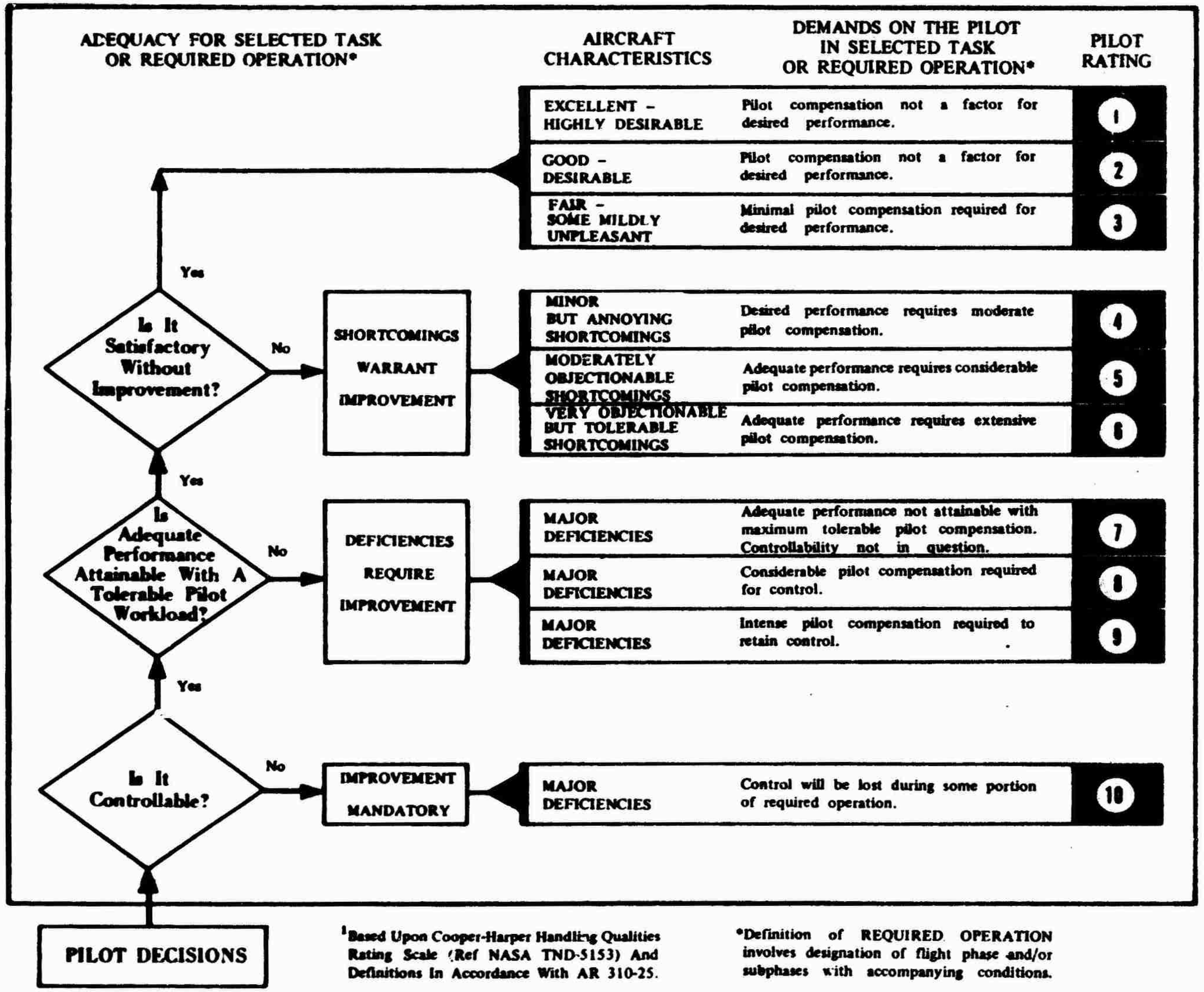
11. A Handling Qualities Rating Scale (HQRS) was used to augment pilot comments and is presented as figure A. The Vibration Rating Scale (VRS) was used to augment pilot comments on vibrations and is presented in figure B.

Deficiency

12. A deficiency is defined as a defect or malfunction discovered during the life cycle of an item of equipment that constitutes a safety hazard to personnel; will result in serious damage to the equipment if operation is continued; or indicates improper design or other cause of failure of an item or part, which seriously impairs the equipment's operational capability.

Shortcoming

13. A shortcoming is defined as an imperfection or malfunction occurring during the life cycle of equipment which must be reported and which should be corrected to increase efficiency and to render the equipment completely serviceable. It will not cause an immediate breakdown, jeopardize safe operation, or materially reduce the usability of the material or end product.



[1]Based Upon Cooper-Harper Handling Qualities Rating Scale (Ref NASA TND-5153) And Definitions In Accordance With AR 310-25.

*Definition of REQUIRED OPERATION involves designation of flight phase and/or subphases with accompanying conditions.

Figure A. Handling Qualities Rating Scale

DEGREE OF VIBRATION	DESCRIPTION[1]	PILOT RATING
No vibration		0
Slight	Not apparent to experienced aircrew fully occupied by their tasks, but noticeable if their attention is directed to it or if not otherwise occupied.	1 2 3
Moderate	Experienced aircrew are aware of the vibration but it does not affect their work, at least over a short period.	4 5 6
Severe	Vibration is immediately apparent to experienced aircrew even when fully occupied. Performance of primary task is affected or tasks can only be done with difficulty.	7 8 9
Intolerable	Sole preoccupation of aircrew is to reduce vibration level.	10

[1] Based upon the Subjective Vibration Assessment Scale developed by the Aeroplane and Armament Experimental Establishment, Boscombe Down, England.

Figure B. Vibration Rating Scale

APPENDIX E. TEST DATA

INDEX

FIGURE 1
LIMITS OF CYCLIC CONTROL TRAVEL
OH-58D USA S/N 69-16285

NOTES: 1. ROTORS STATIC
2. CONTROL POSITIONS MEASURED AT CENTER OF GRIP
3. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY EXTERNAL POWER SOURCES
4. HYDRAULIC AND FORCE TRIM SYSTEMS ON
5. COLLECTIVE CONTROL FULL DOWN

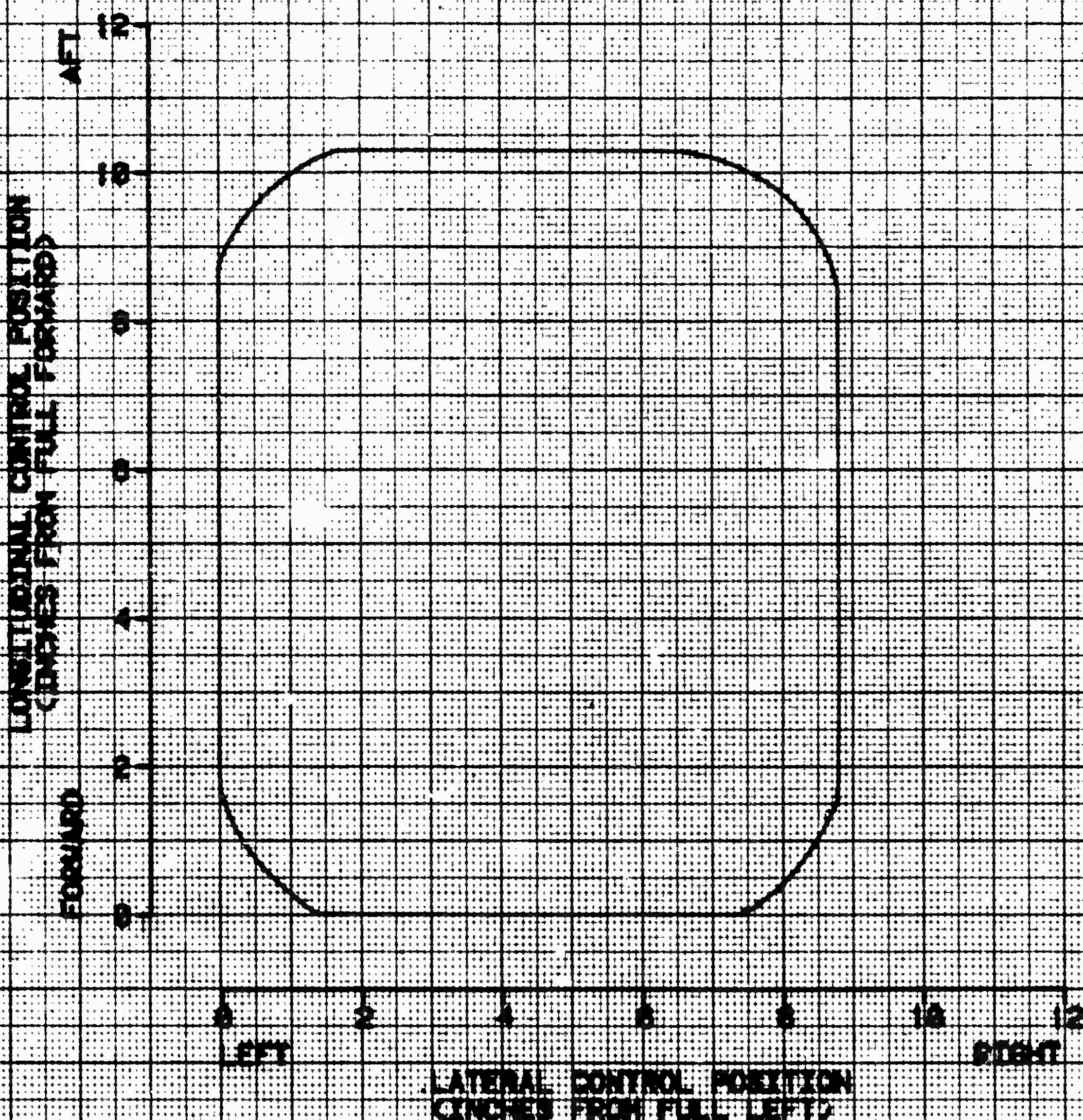


FIGURE 2
LONGITUDINAL CONTROL SYSTEM CHARACTERISTICS
OH-58D USA S/N 89-16285

NOTES:
1. ROTORS STATIC
2. FORCES AND POSITIONS MEASURED AT CENTER OF CONTROL GRIP
3. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY EXTERNAL POWER SOURCES
4. HYDRAULIC AND FORCE TRIM SYSTEMS ON
5. TOTAL LONGITUDINAL CONTROL TRAVEL = 10.3 INCHES

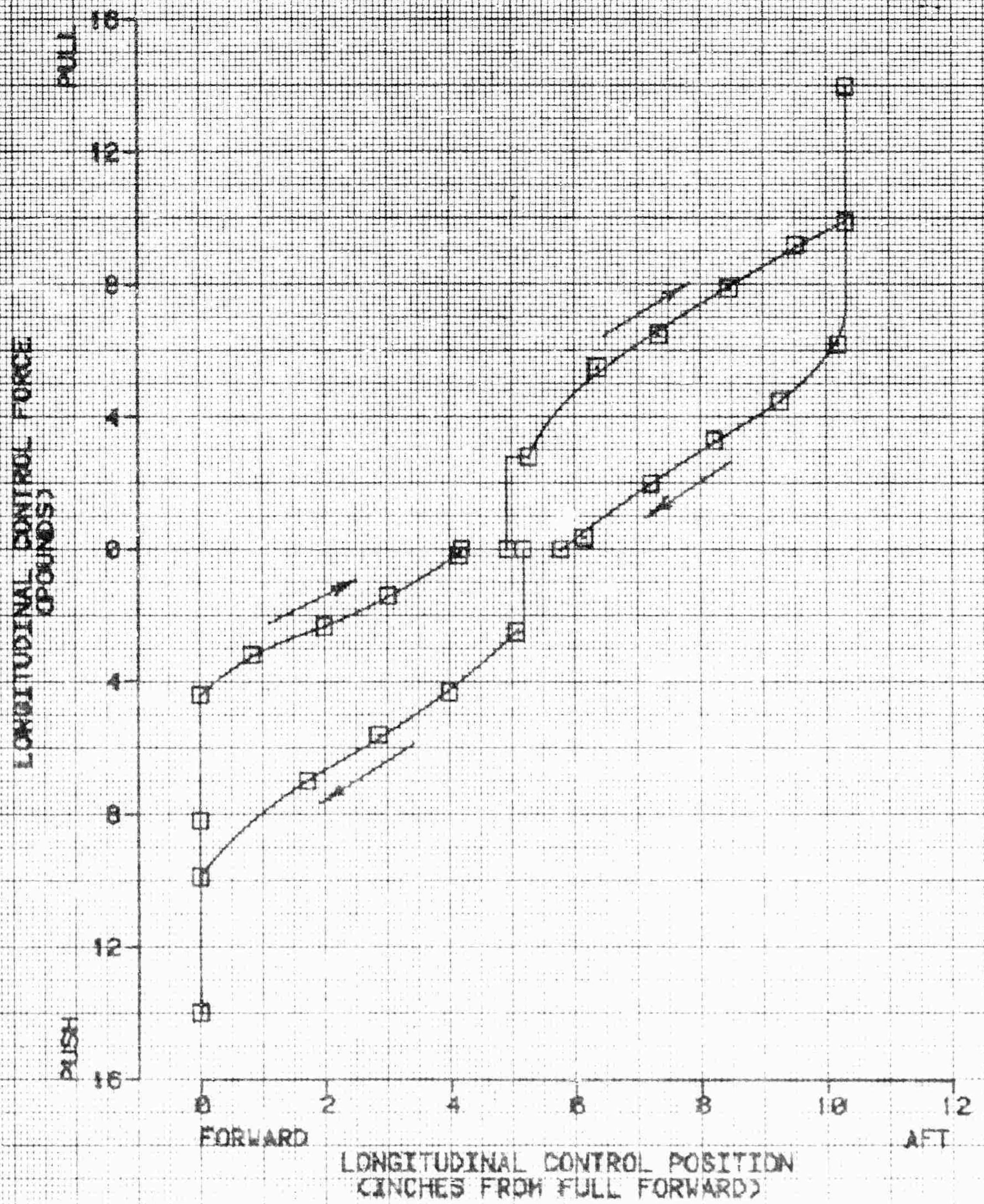


FIGURE 3
LATERAL CONTROL SYSTEM CHARACTERISTICS
OH-58D USA S/N 69-16285

NOTES:
1. ROTORS STATIC
2. FORCES AND POSITIONS MEASURED AT CENTER OF CONTROL GRIP
3. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY EXTERNAL POWER SOURCES
4. HYDRAULIC AND FORCE TRIM SYSTEMS ON
5. TOTAL LATERAL CONTROL TRAVEL = 8.8 INCHES

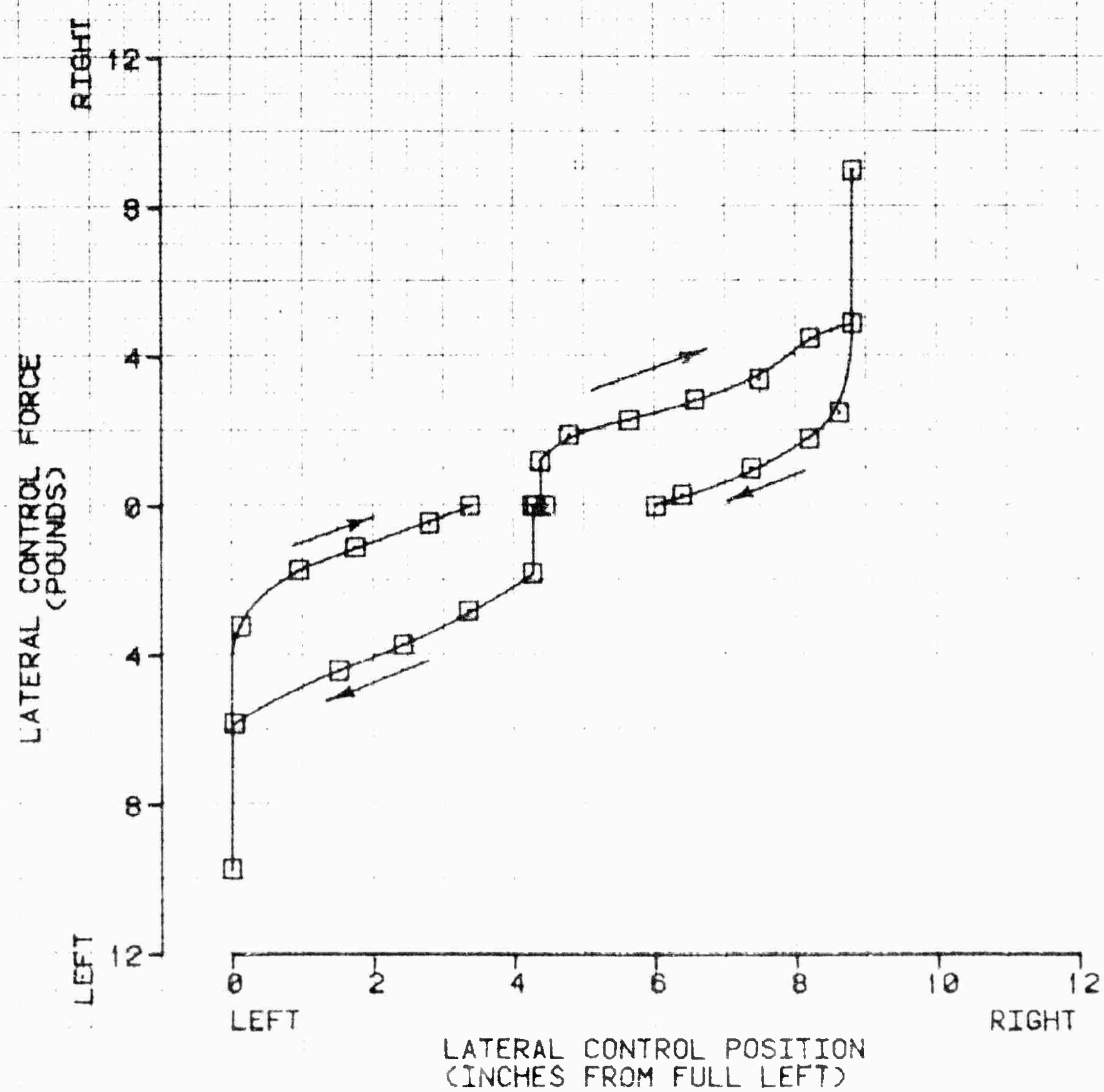


FIGURE 4
DIRECTIONAL CONTROL SYSTEM CHARACTERISTICS
OH-58D USA S/N 69-16285

NOTES: 1. ROTORS STATIC
2. FORCES AND POSITIONS MEASURED AT CENTER OF PEDAL
3. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY EXTERNAL POWER SOURCES
4. HYDRAULIC AND FORCE TRIM SYSTEMS ON
5. TOTAL DIRECTIONAL CONTROL TRAVEL = 6.8 INCHES

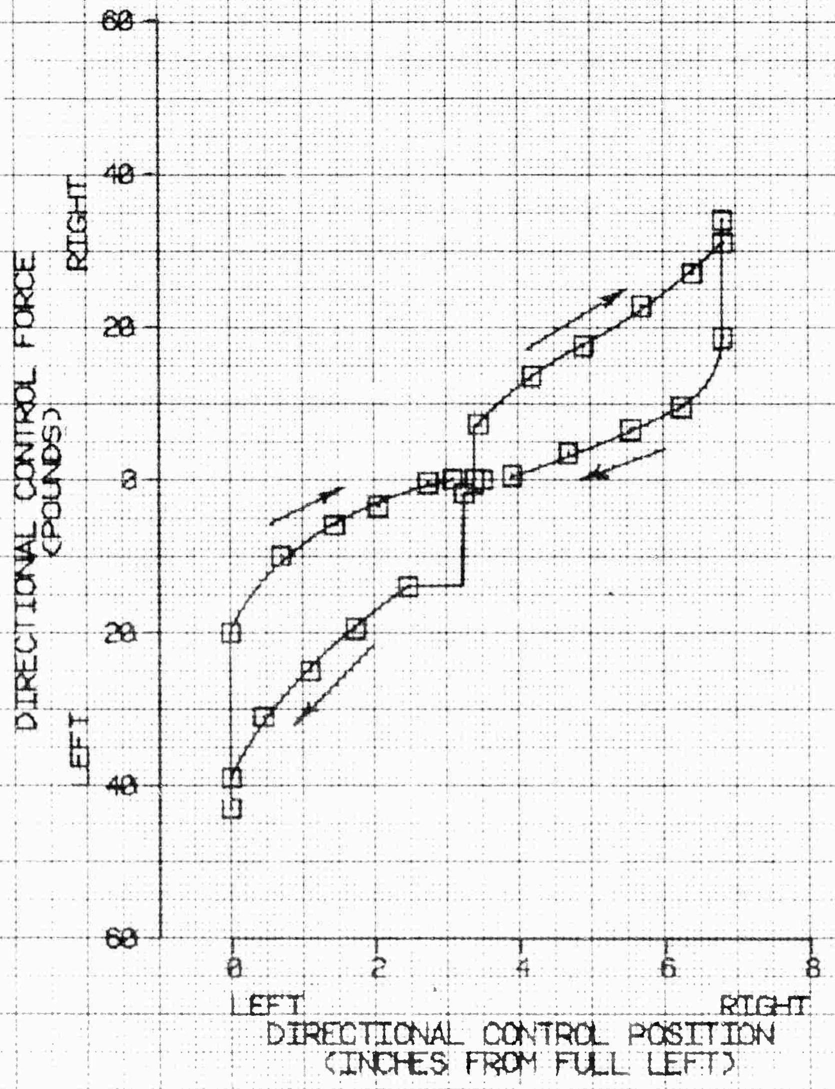


FIGURE 5
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
4230	109.9 (AFT)	2960	19.5	396	LEVEL

NOTES:
1. SCAS ON
2. EXTENDED STABILIZER INSTALLED
3. SHADED SYMBOLS DENOTE DIVES
4. ZERO SIDESLIP

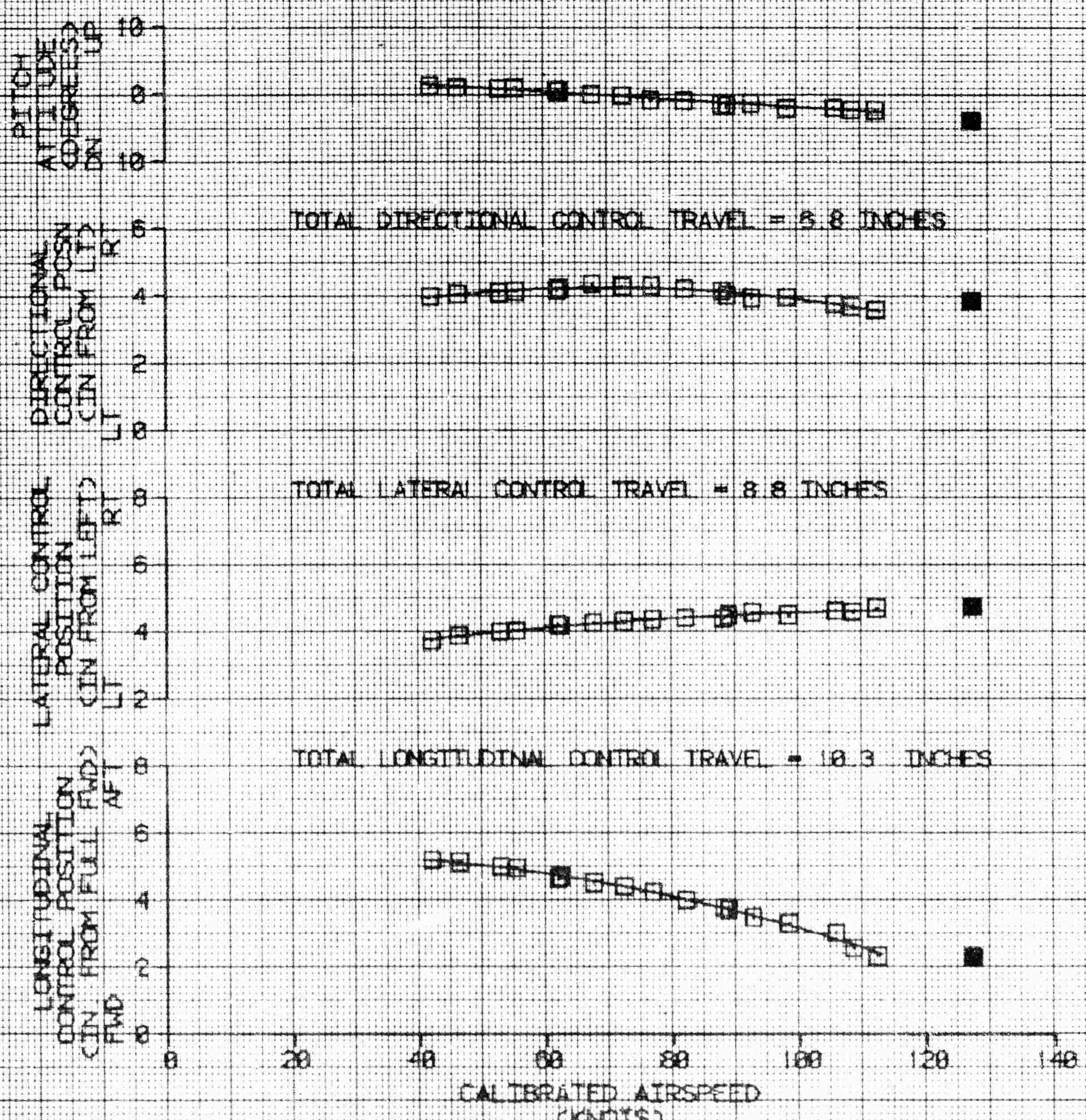


FIGURE 6
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
4420	109.9 (AFT)	10430	11.0	397	LEVEL

NOTES: 1. SCAS ON
2. EXTENDED STABILIZER INSTALLED
3. ZERO SIDESLIP

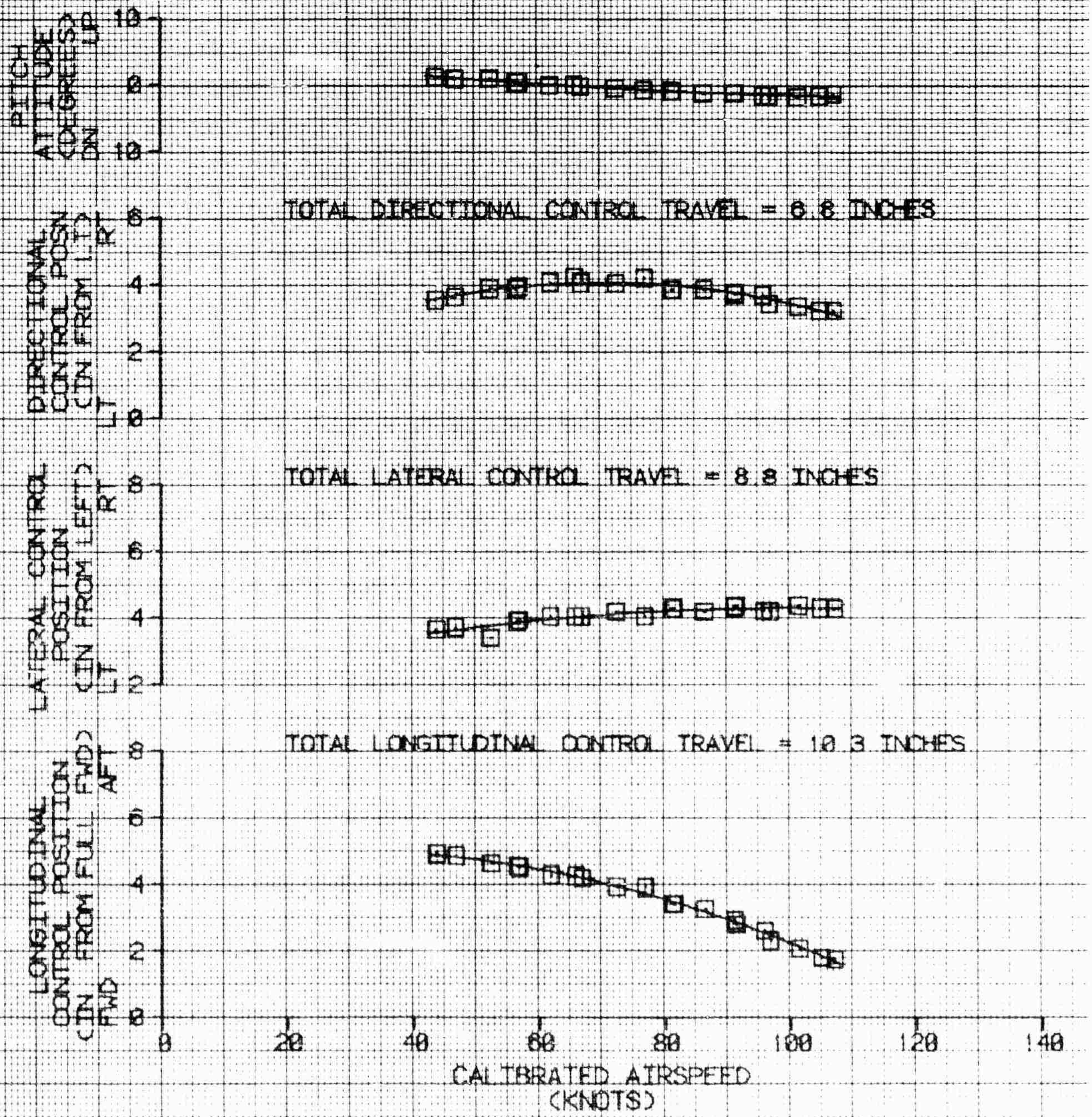


FIGURE 7
STATIC LONGITUDINAL STABILITY
OH-58D USA S/N 69-16285

SYM	AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
⊡	3920	112.8 (AFT)	6540	16.0	395	CLIMB
△	3830	112.8 (AFT)	5930	16.5	401	AUTO

NOTES: 1. SHADED SYMBOLS DENOTE TRIM
2. SCAS ON
3. MAST MOUNTED SIGHT REMOVED
4. SMALL STABILIZER INSTALLED

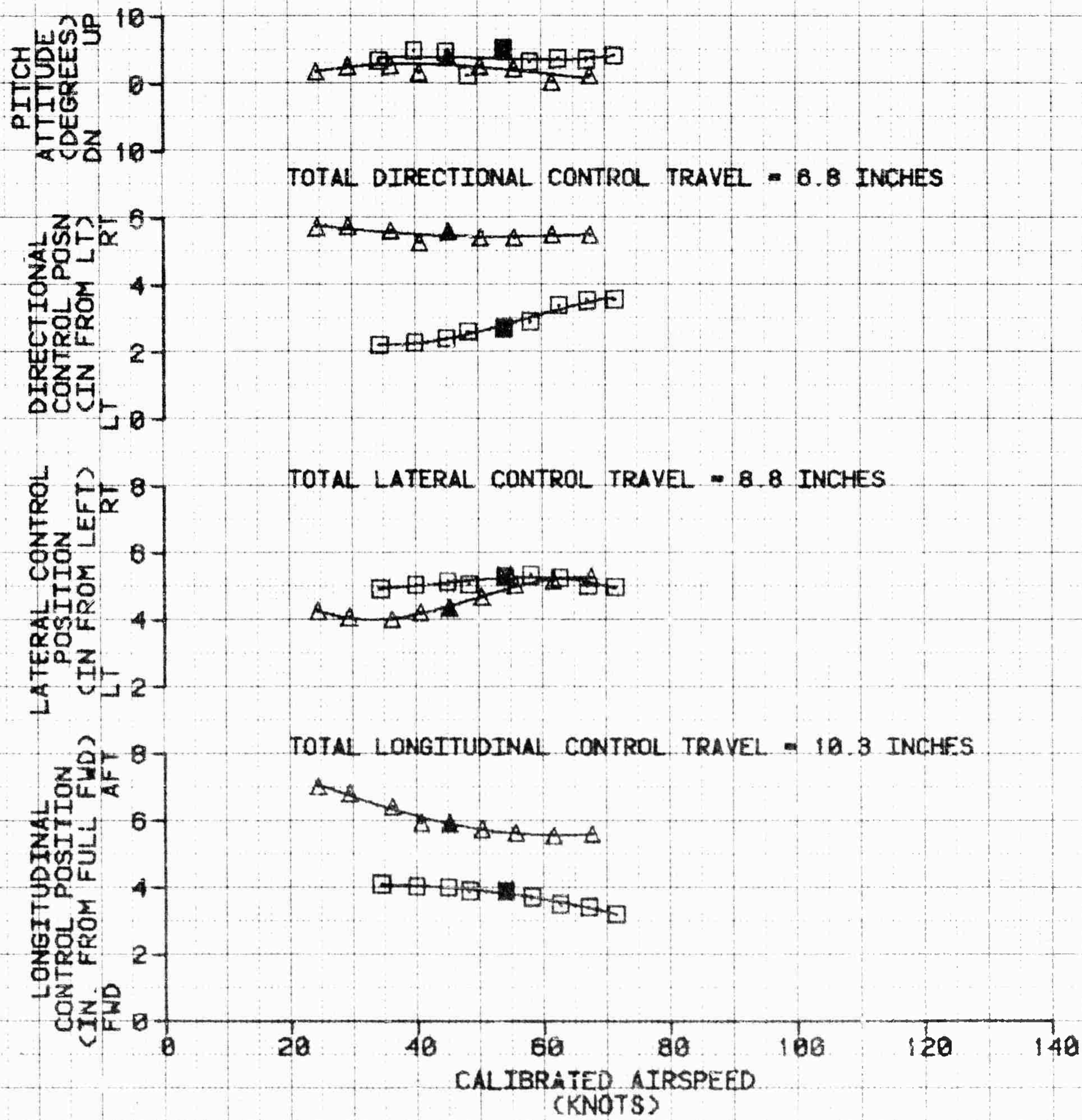


FIGURE 8
STATIC LONGITUDINAL STABILITY
OH-58D USA S/N 69-16285

SYM	AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
⊡	3890	112.5 (AFT)	4880	17.5	395	LEVEL
△	3800	112.5 (AFT)	6060	16.5	395	LEVEL

NOTES:
1. SHADED SYMBOLS DENOTE TRIM
2. SCAS ON
3. MAST MOUNTED SIGHT REMOVED
4. SMALL STABILIZER INSTALLED

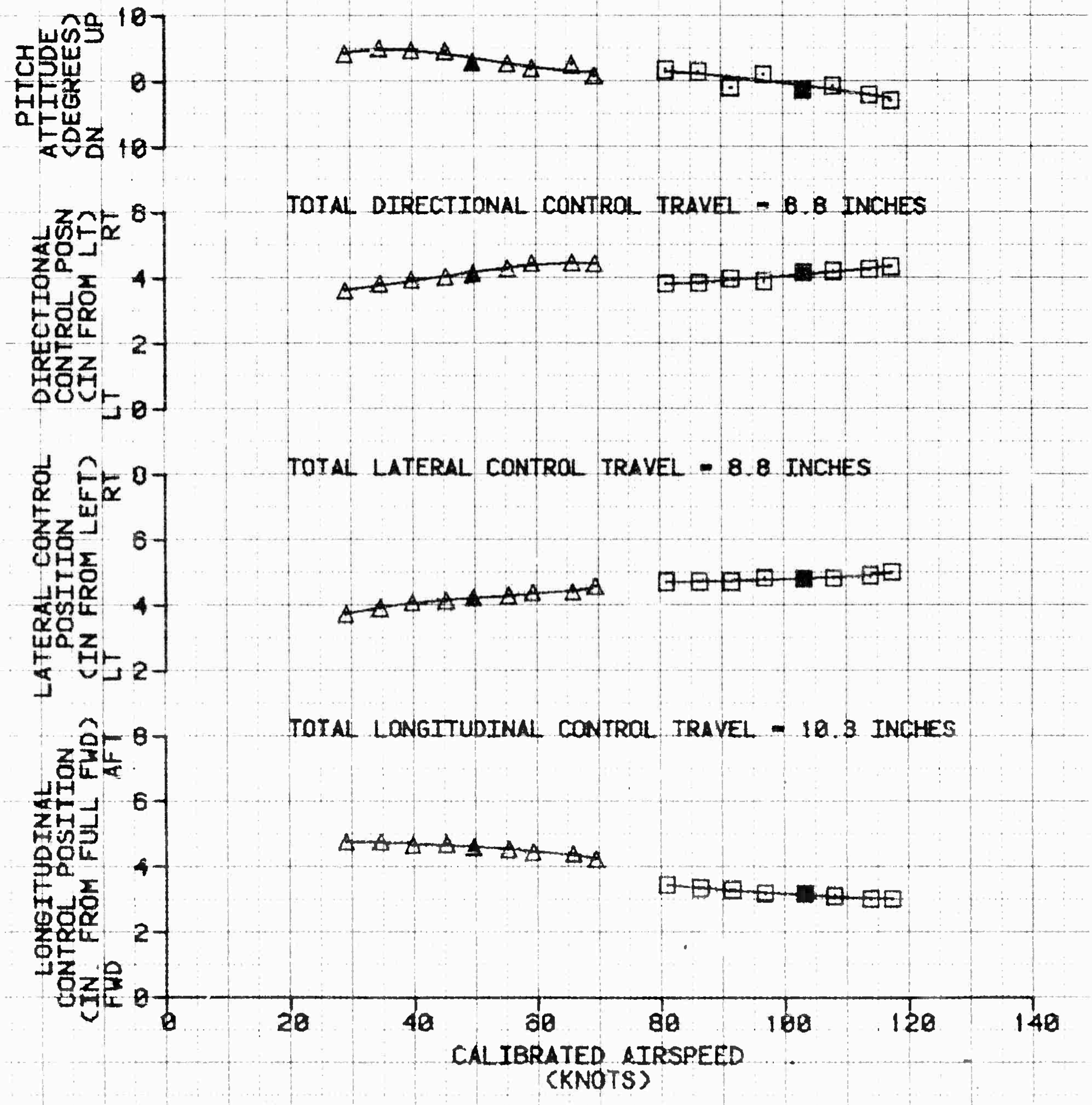


FIGURE 9

STATIC LONGITUDINAL STABILITY

OH-58D USA S/N 89-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
4140	111.1 (AFT)	8070	9.0	395	CLIMB

NOTES: 1. SHADED SYMBOLS DENOTE TRIM
2. SCAS ON
3. SMALL STABILIZER INSTALLED

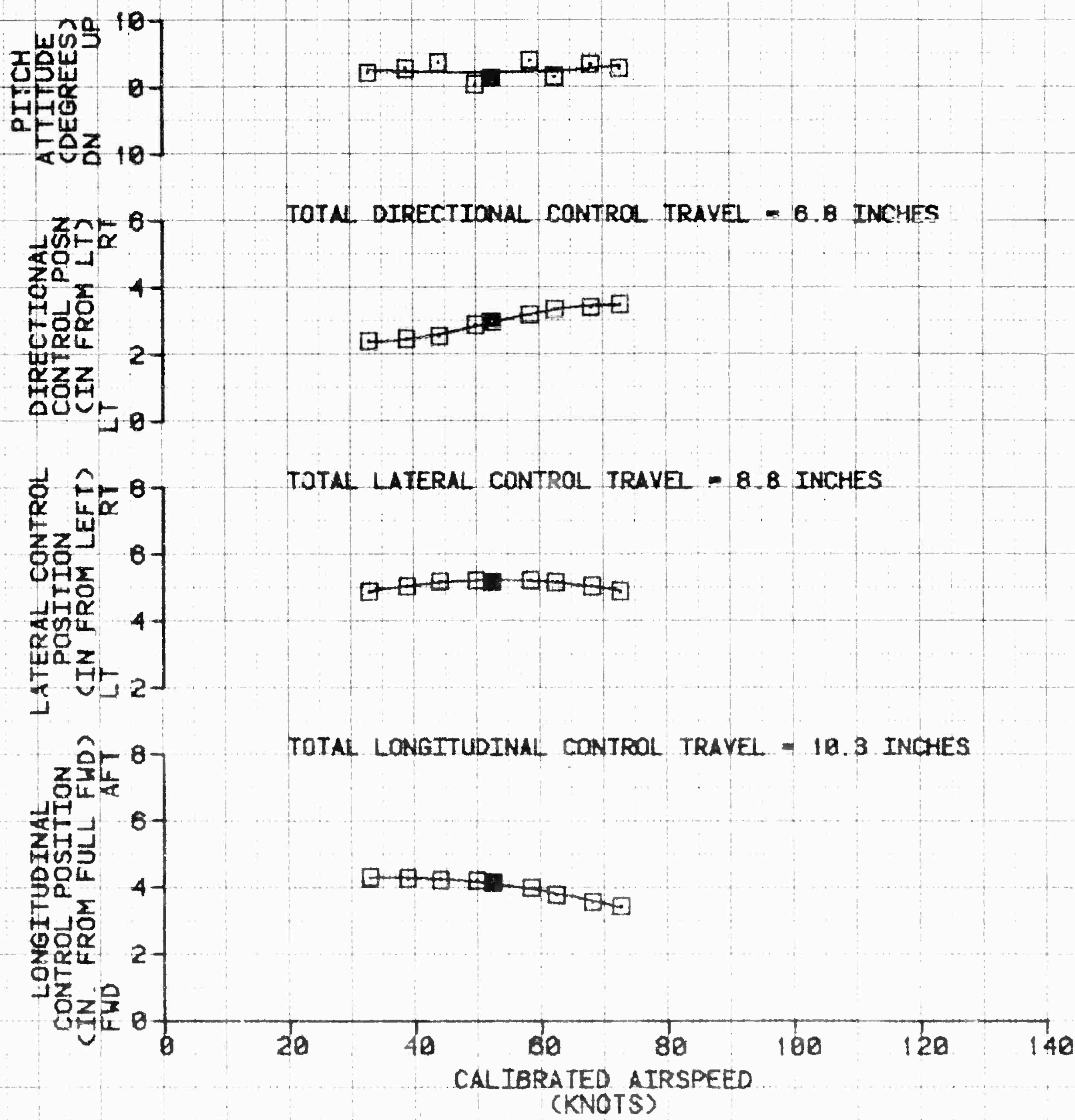


FIGURE 10

STATIC LONGITUDINAL STABILITY
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
4250	111.0 (AFT)	5730	8.0	395	AUTO

NOTES: 1. SHADED SYMBOLS DENOTE TRIM
2. SCAS ON
3. SMALL STABILIZER INSTALLED

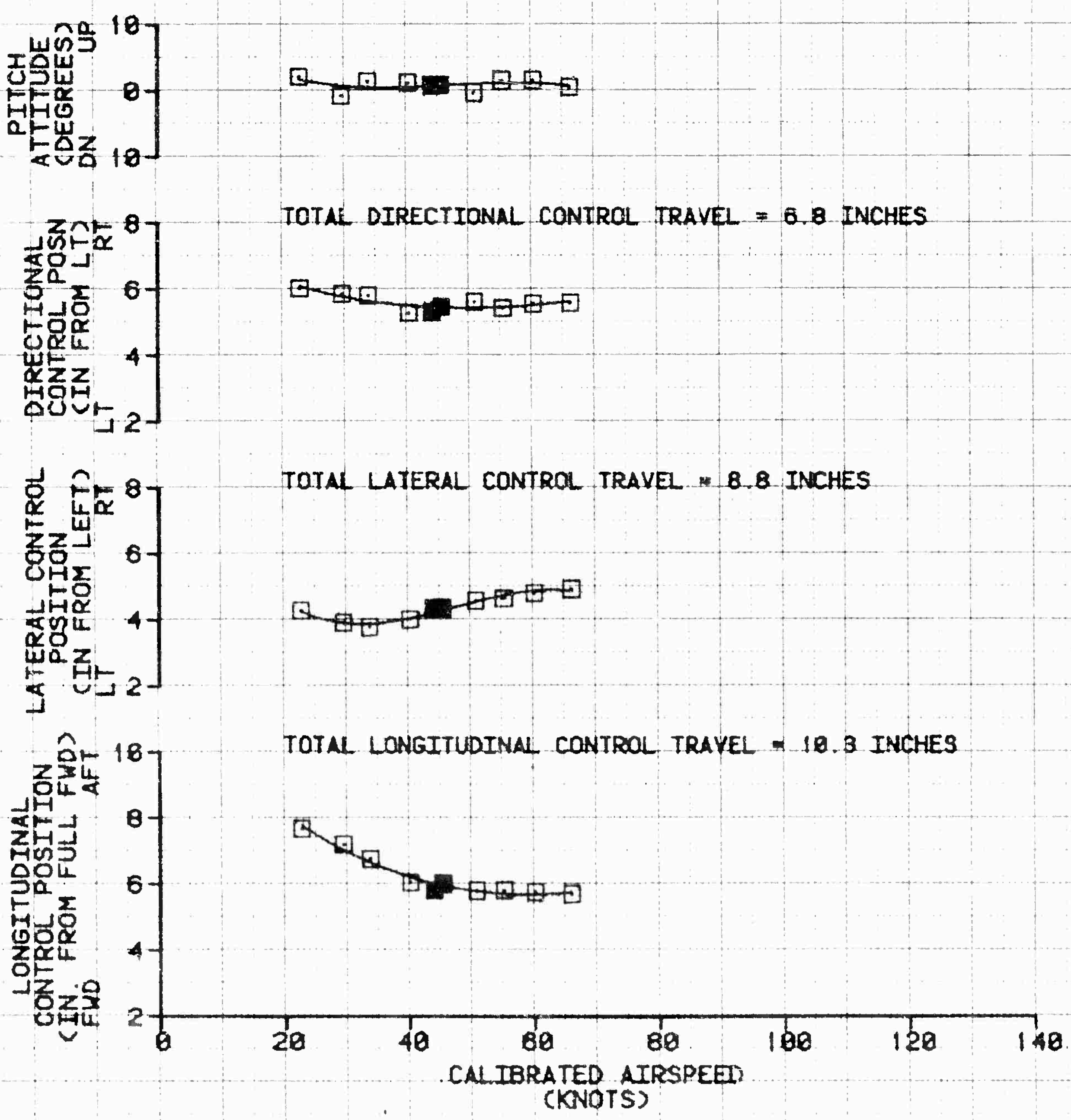


FIGURE 11
STATIC LONGITUDINAL STABILITY
OH-58D USA S/N 89-16285

SYM	AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
⊡	4220	111.1 (AFT)	5550	8.5	395	LEVEL
△	4210	111.1 (AFT)	6010	8.5	395	LEVEL

NOTES: 1. SHADED SYMBOLS DENOTE TRIM
2. SCAS ON
3. SMALL STABILIZER INSTALLED

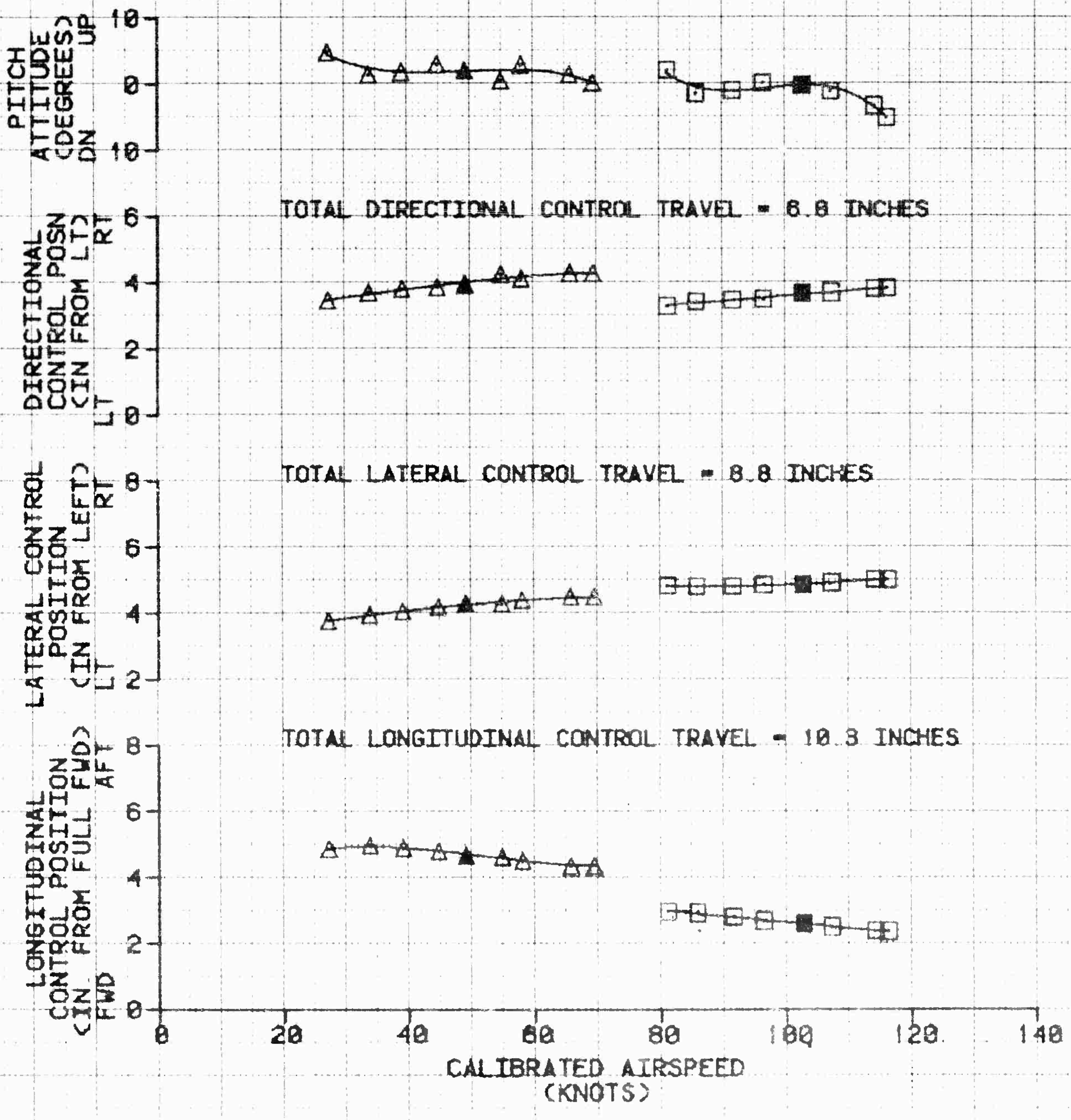


FIGURE 12

STATIC LONGITUDINAL STABILITY
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
4230	111.0 (AFT)	6650	17.0	395	CLIMB

NOTES:
1. SHADED SYMBOLS DENOTE TRIM
2. SCAS ON
3. EXTENDED STABILIZER INSTALLED

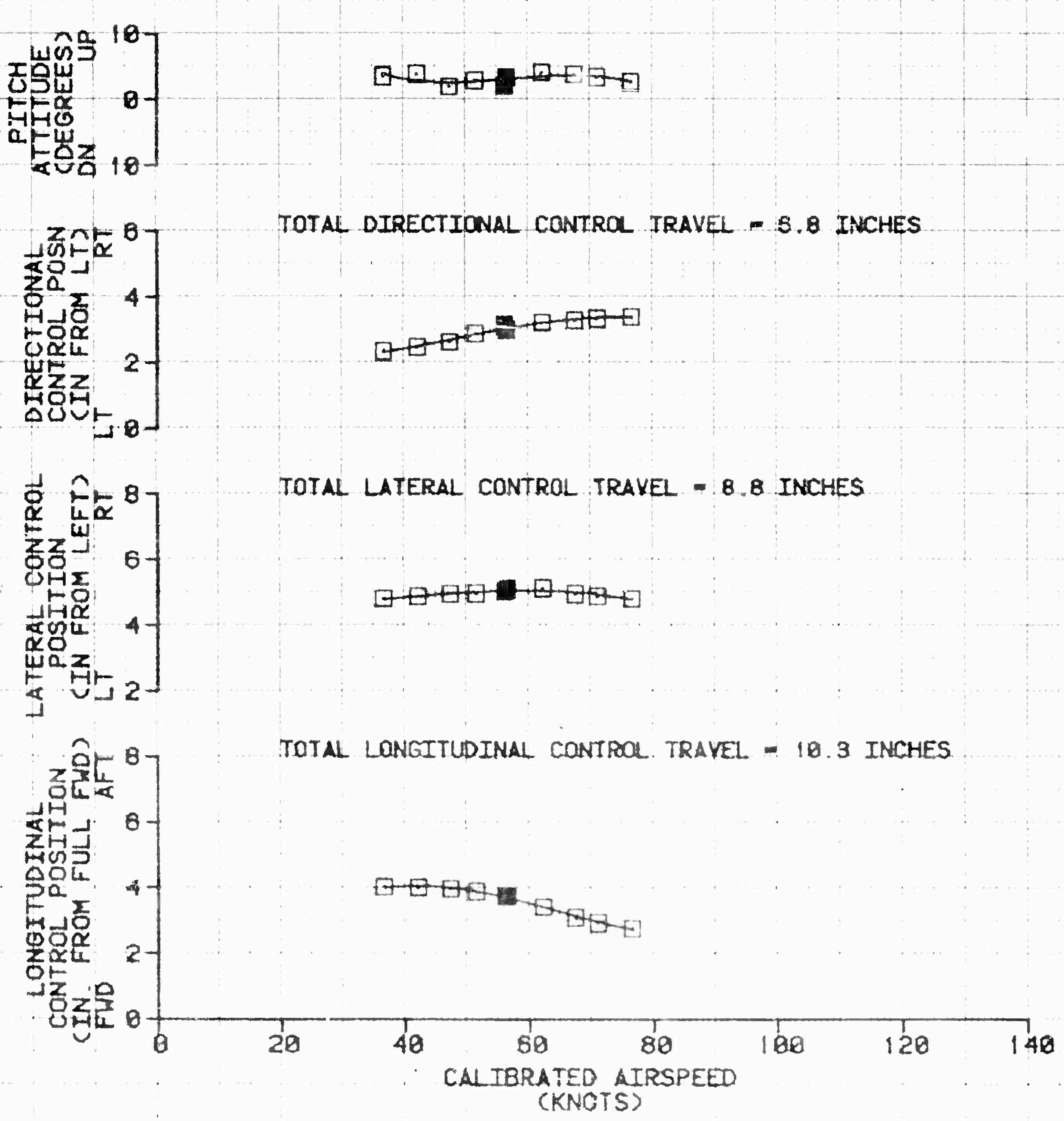


FIGURE 13

STATIC LONGITUDINAL STABILITY

OH-58D USA S/N 69-16285

SYM	AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
⊡	4280	111.0 (AFT)	6870	18.6	395	LEVEL
△	4160	111.0 (AFT)	5700	16.5	395	LEVEL

NOTES: 1. SHADED SYMBOLS DENOTE TRIM
2. SCAS ON
3. EXTENDED STABILIZER INSTALLED

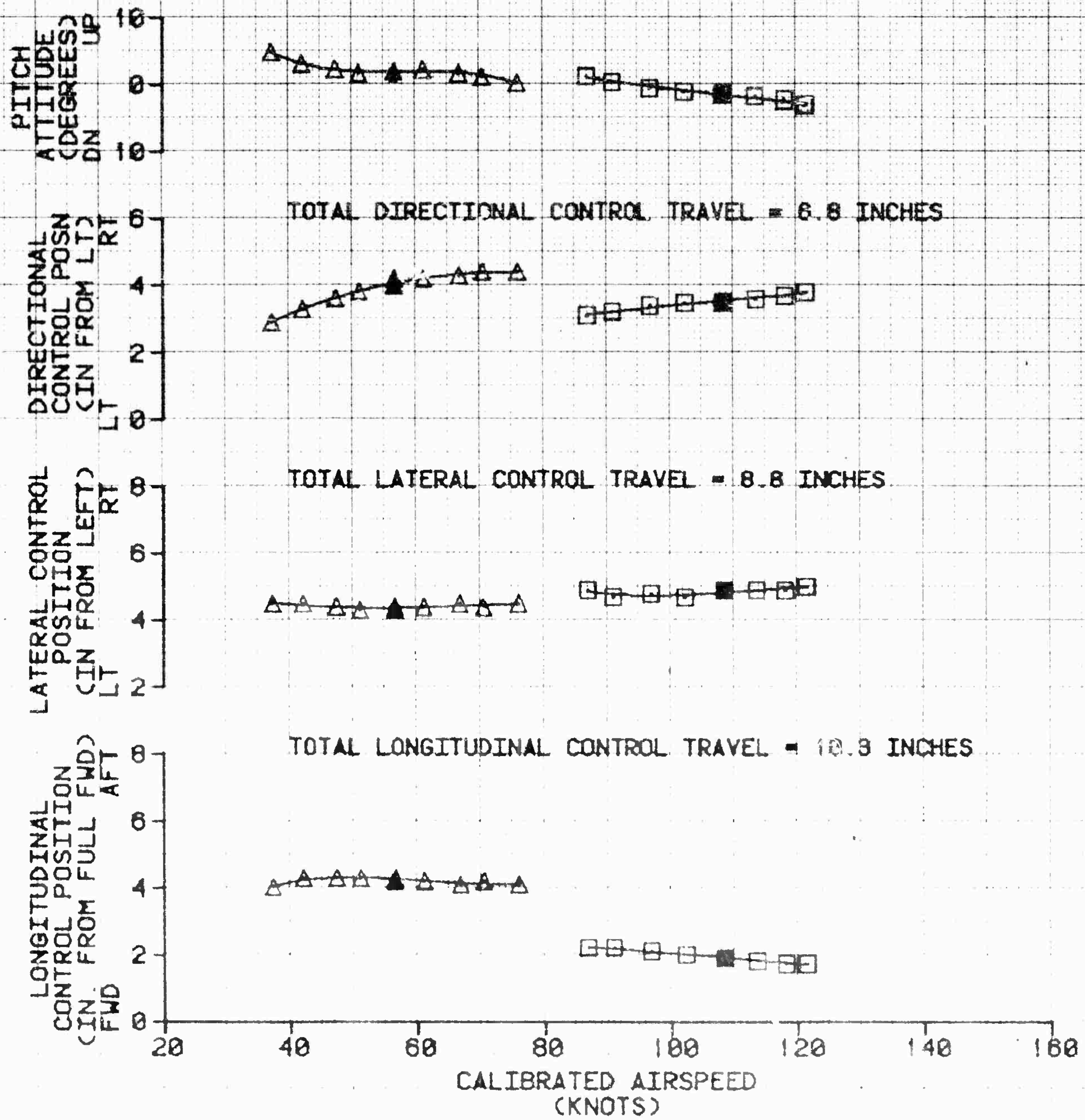


FIGURE 14
STATIC LATERAL-DIRECTIONAL STABILITY
UH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4180	111.1 (AFT)	6360	8.0	395	40	LEVEL

NOTES: 1. SHADED SYMBOLS DENOTE TRIM
2. SCAS ON
3. SMALL STABILIZER INSTALLED

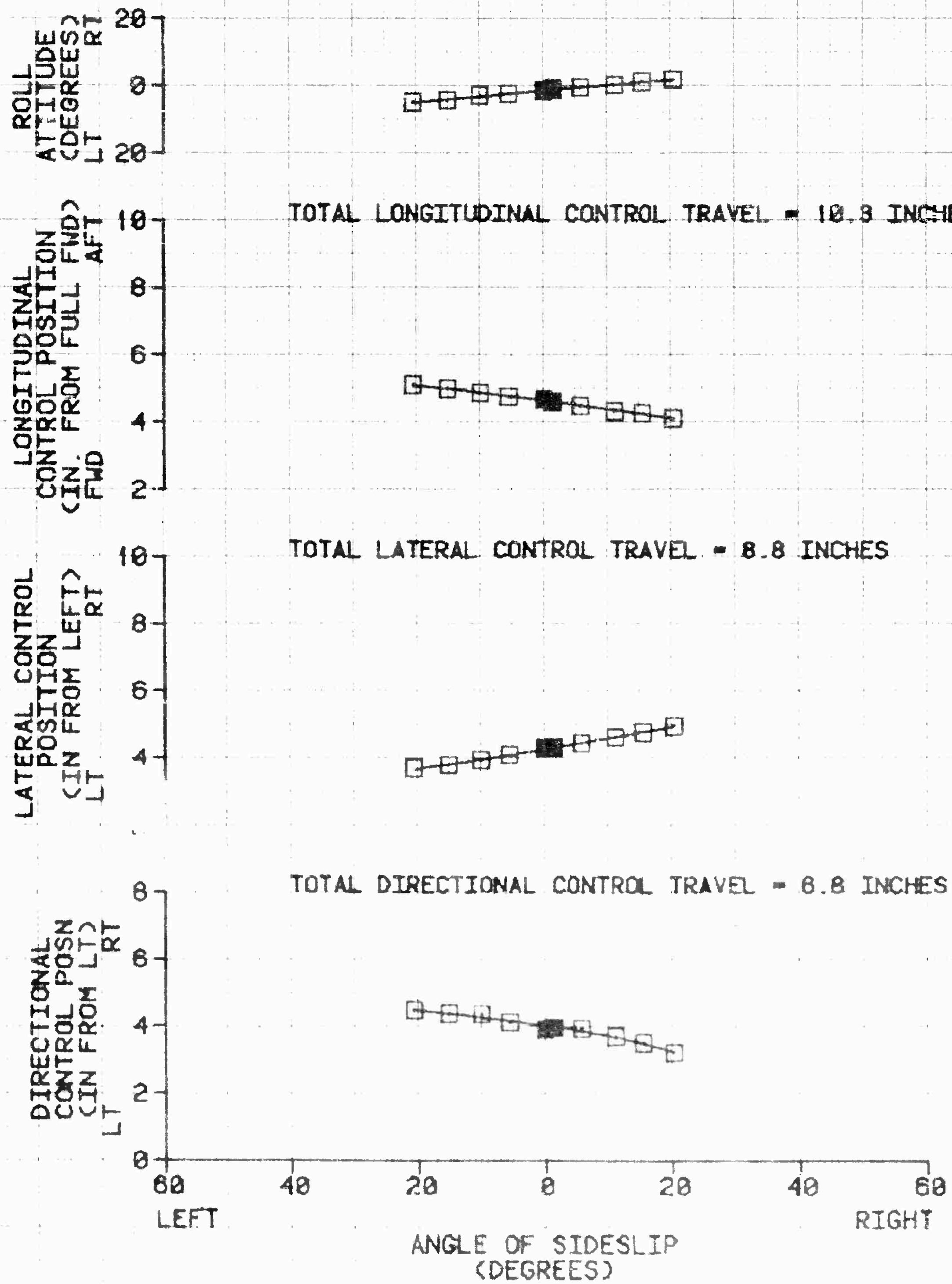


FIGURE 15
STATIC LATERAL-DIRECTIONAL STABILITY
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4230	111.0 (AFT)	7060	10.9	395	107	LEVEL

NOTES: 1. SHADED SYMBOLS DENOTE TRIM
2. SCAS ON
3. EXTENDED STABILIZER INSTALLED

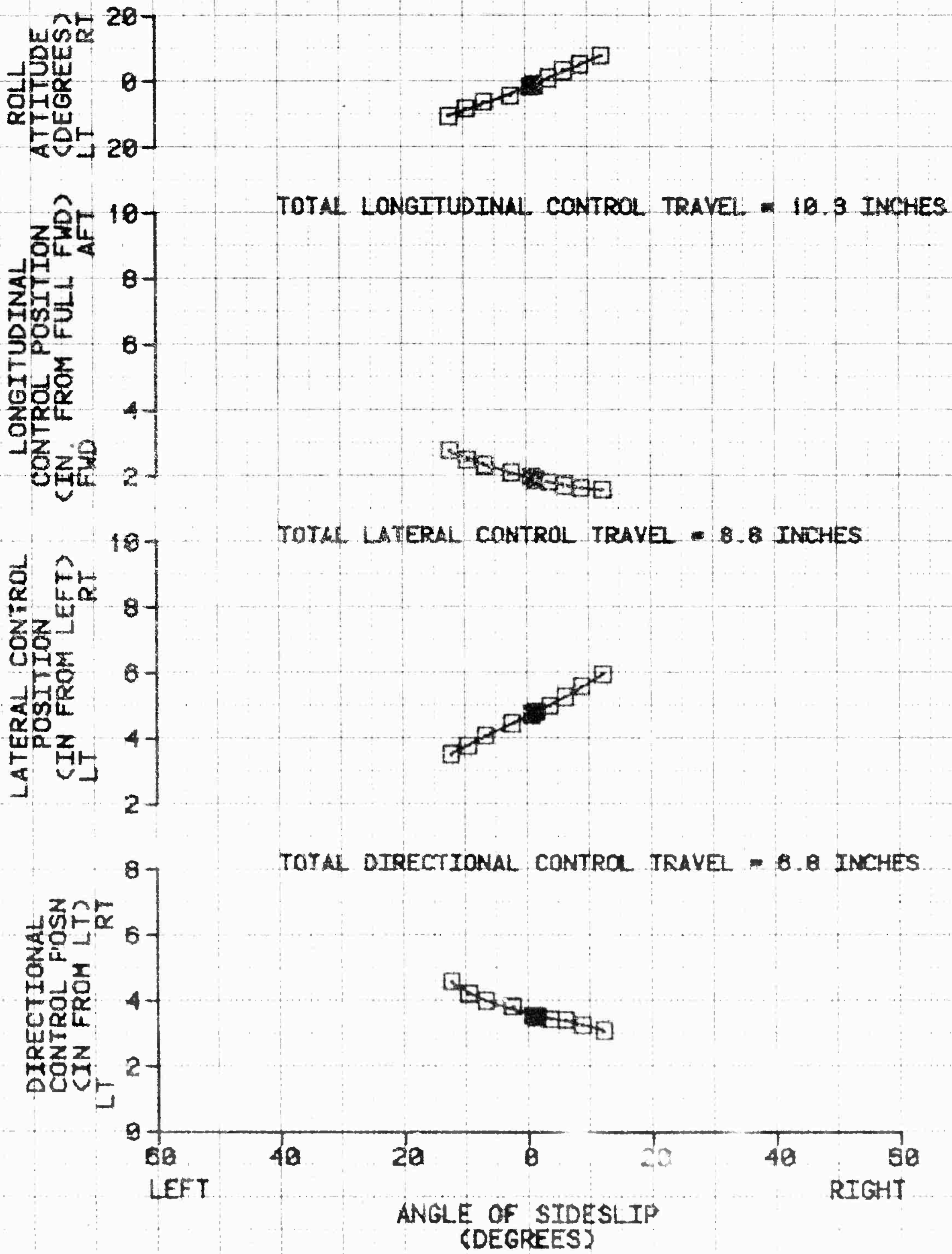


FIGURE 16

STATIC LATERAL-DIRECTIONAL STABILITY

OH-58D USA S/N 69-16295

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4250	111.2 (AFT)	6180	10.0	395	102	LEVEL

NOTES: 1. SHADED SYMBOLS DENOTE TRIM
2. SCAS ON
3. SMALL STABILIZER INSTALLED

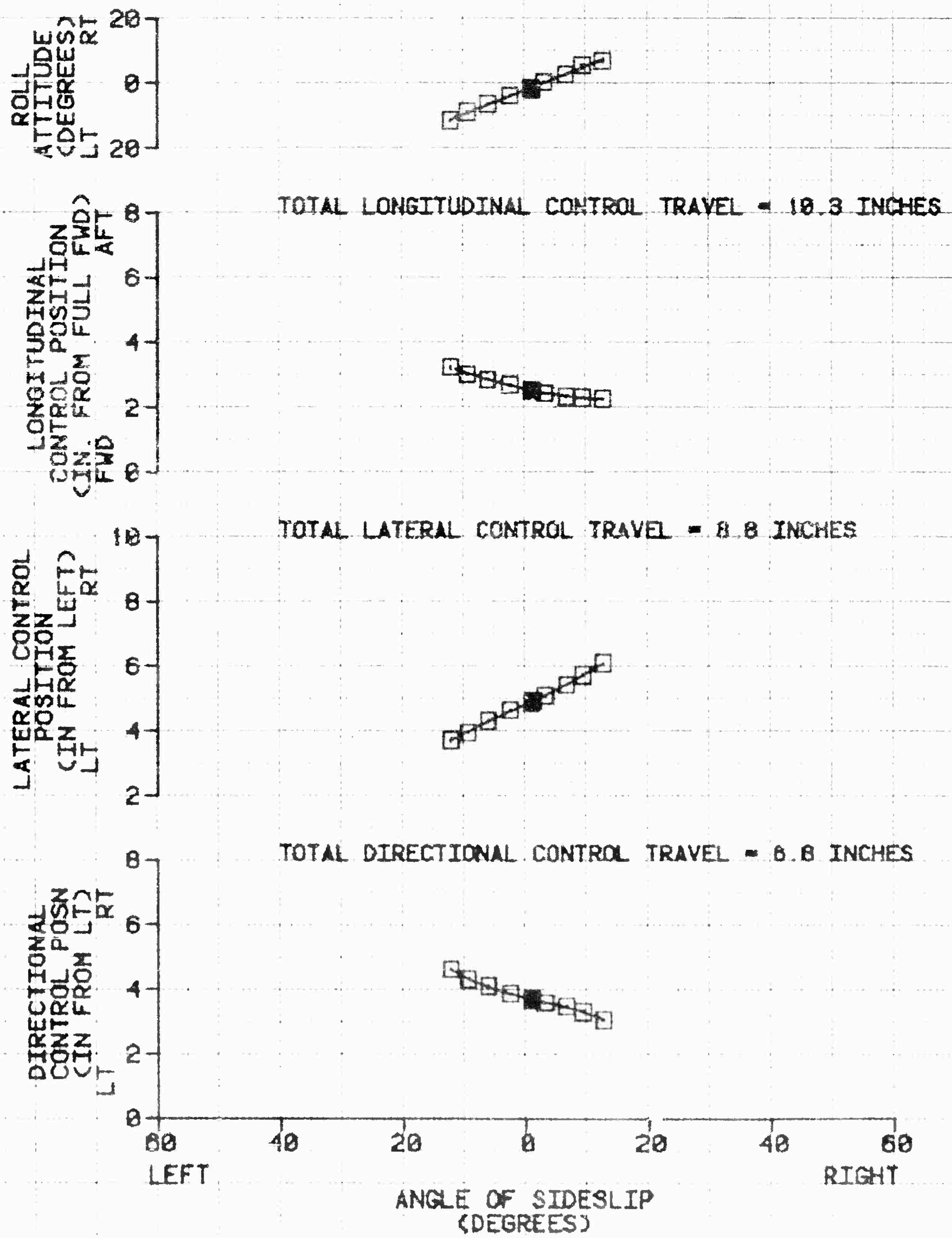


FIGURE 17

STATIC LATERAL-DIRECTIONAL STABILITY

OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
3750	112.5 (AFT)	6090	16.5	395	50	LEVEL

NOTES:
1. SHADED SYMBOLS DENOTE TRIM
2. SCAS ON
3. MAST MOUNTED SIGHT REMOVED
4. SMALL STABILIZER INSTALLED

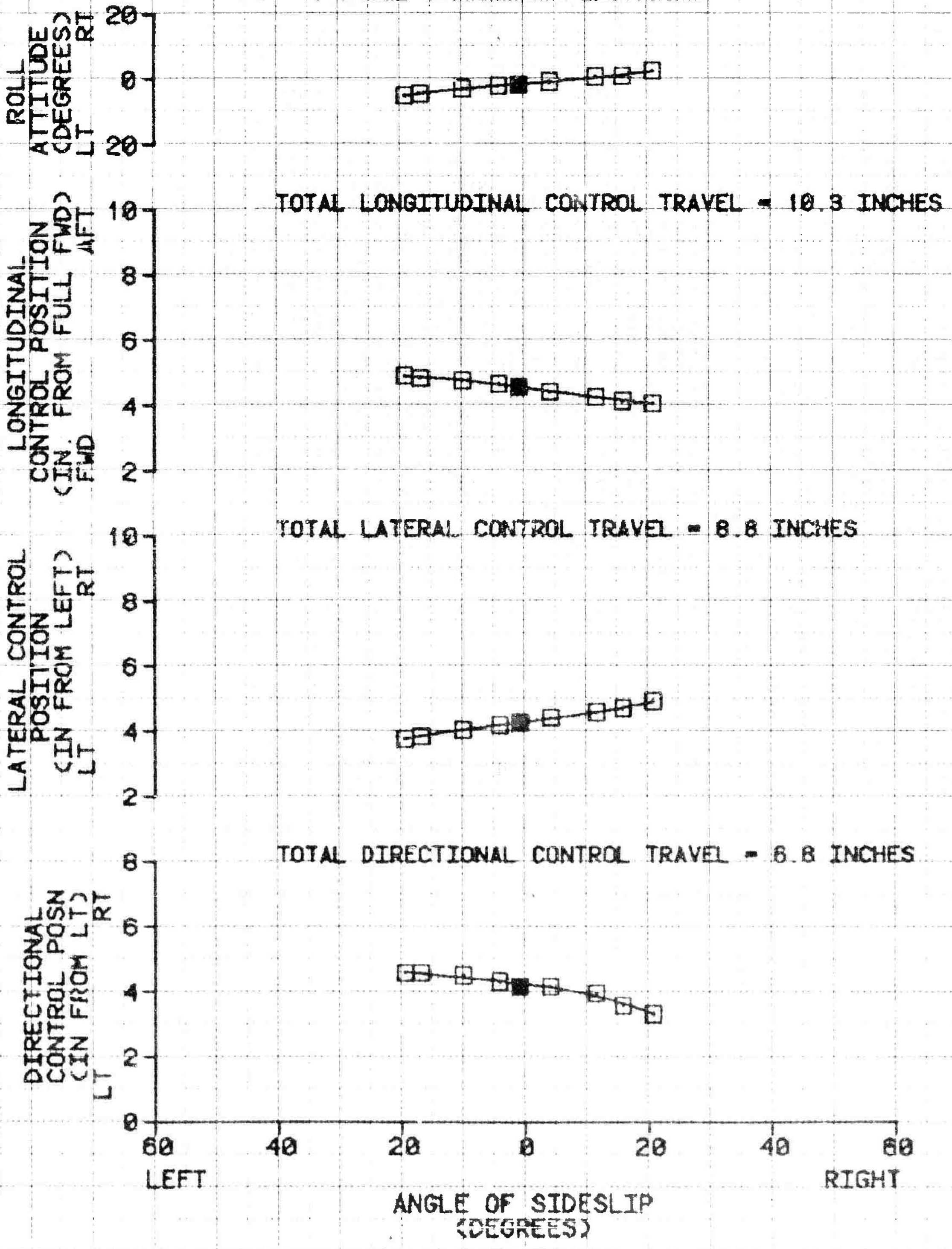


FIGURE 18
STATIC LATERAL-DIRECTIONAL STABILITY
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
3840	112.5 (AFT)	5870	18.5	395	102	LEVEL

NOTES: 1. SHADED SYMBOLS DENOTE TRIM
2. SCAS ON
3. MAST MOUNTED SIGHT REMOVED
4. SMALL STABILIZER INSTALLED

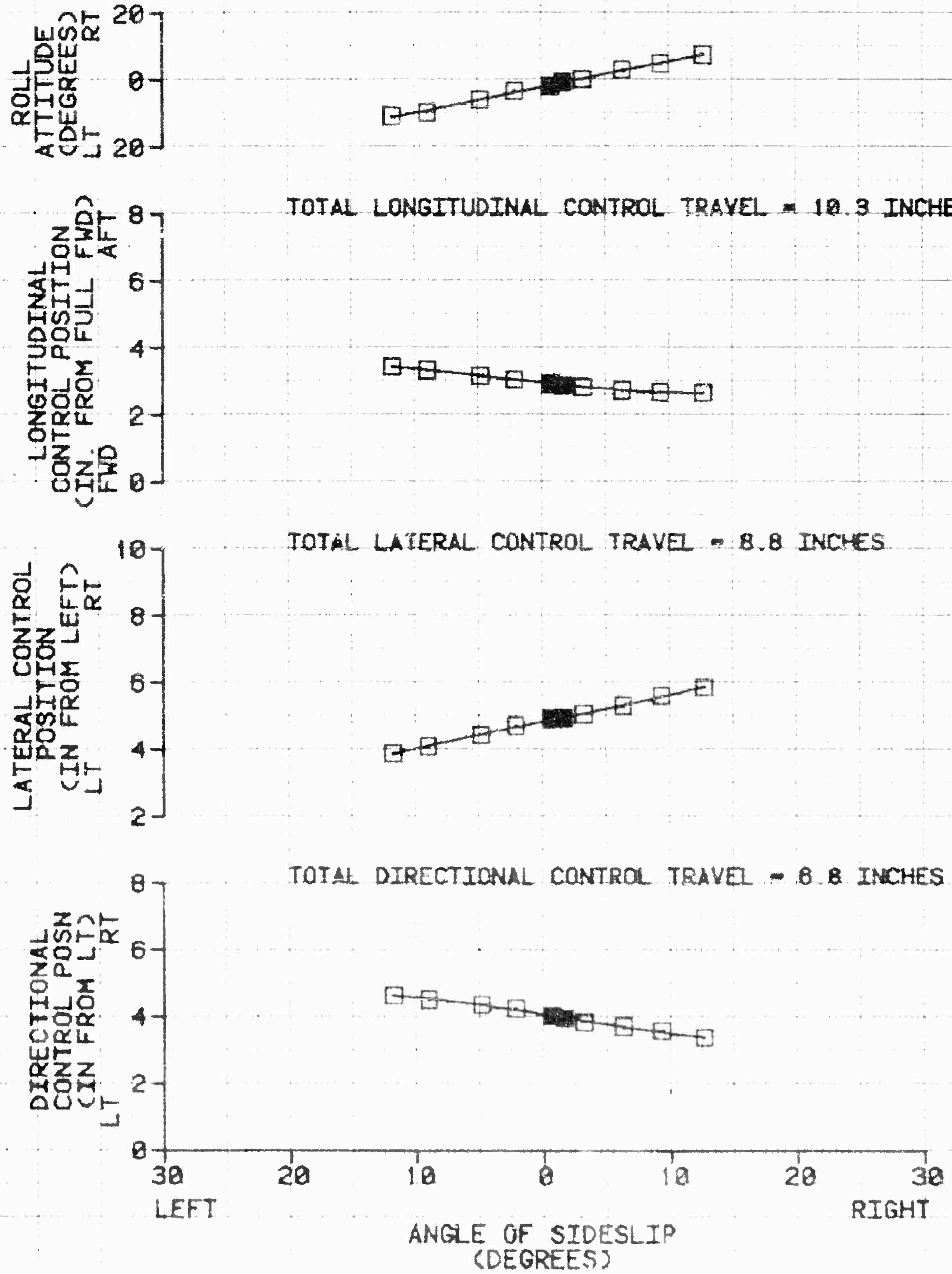


FIGURE 19
STATIC LATERAL-DIRECTIONAL STABILITY
OH-58D USA S/N 69-16285

SYM	AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
⊡	3870	112.6 (AFT)	6630	15.5	395	53	CLIMB
△	3760	112.6 (AFT)	7100	15.0	404	46	AUTO

NOTES:
1. SHADED SYMBOLS DENOTE TRIM
2. SCAS ON
3. MAST MOUNTED SIGHT REMOVED
4. SMALL STABILIZER INSTALLED

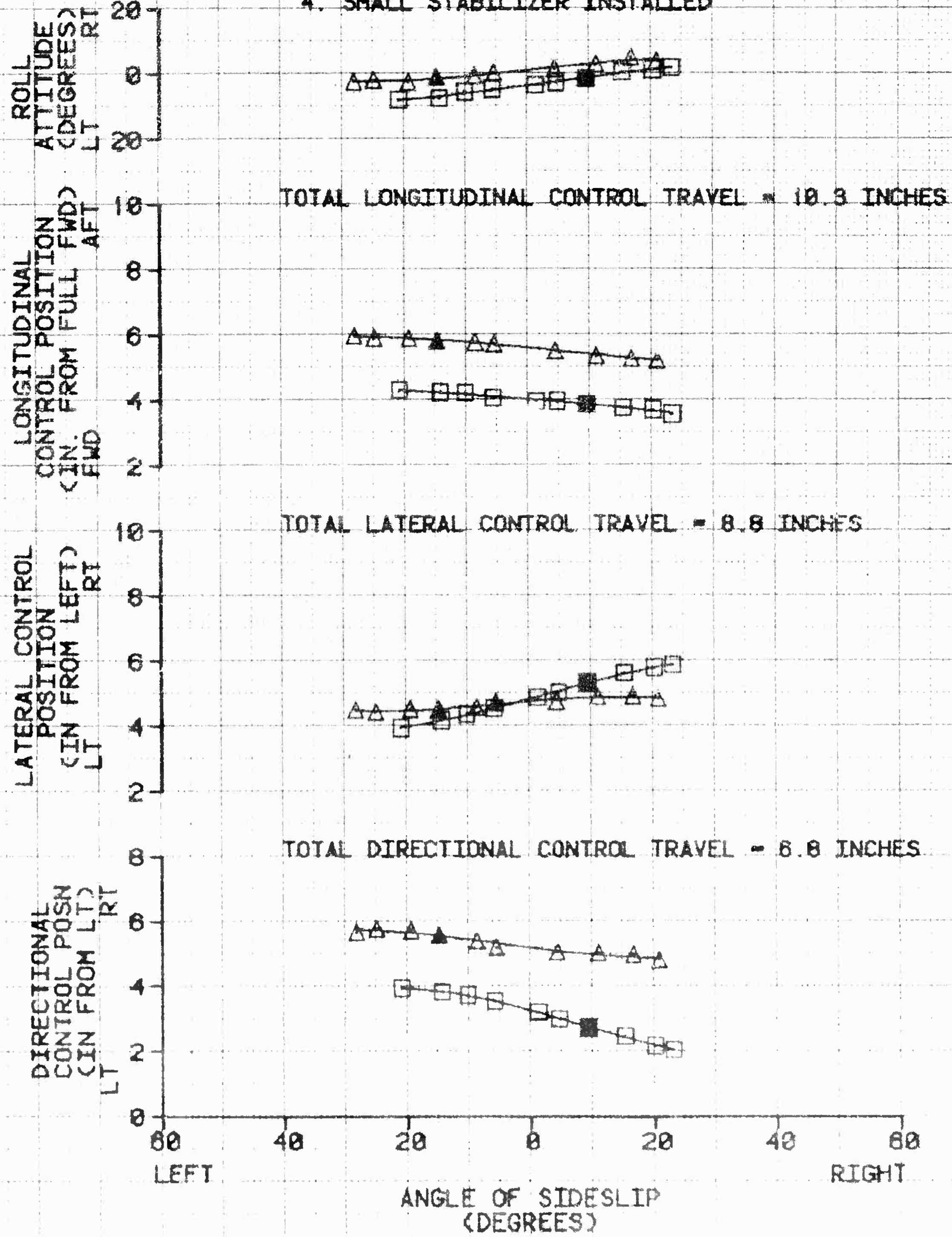


FIGURE 20
STATIC LATERAL-DIRECTIONAL STABILITY
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4280	112.2 (AFT)	8310	12.5	394	40	LEVEL

NOTES: 1. SHADED SYMBOLS DENOTE TRIM
2. SCAS ON
3. SMALL STABILIZER INSTALLED
4. CREW DOORS REMOVED

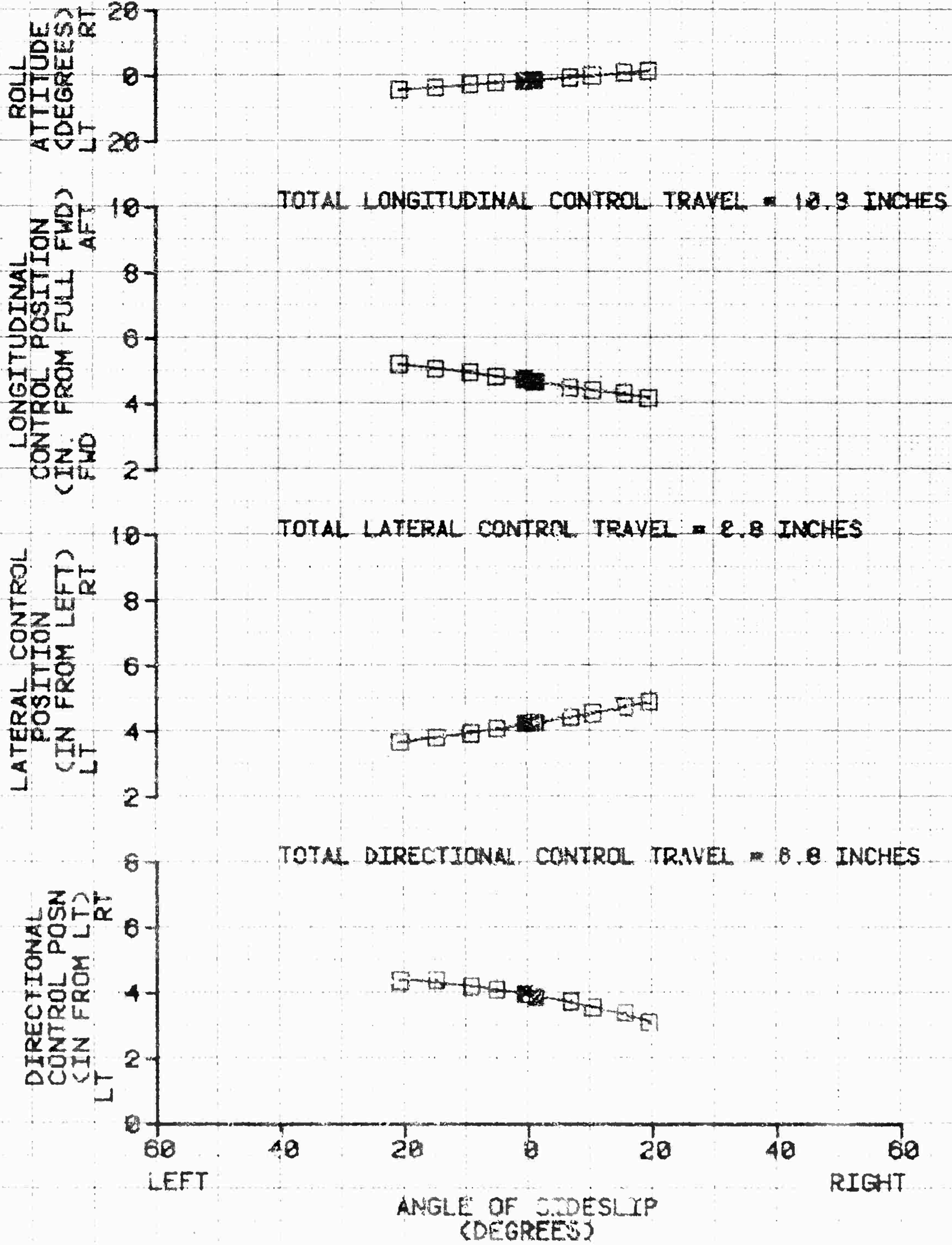


FIGURE 21
MANEUVERING STABILITY
UH-58D USA S/N 69-16265

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4210	111.2 (AFT)	5850	17.5	395	100	LEFT TURN

NOTES: 1. SHADED SYMBOLS DENOTE TRIM
2. SCAS ON

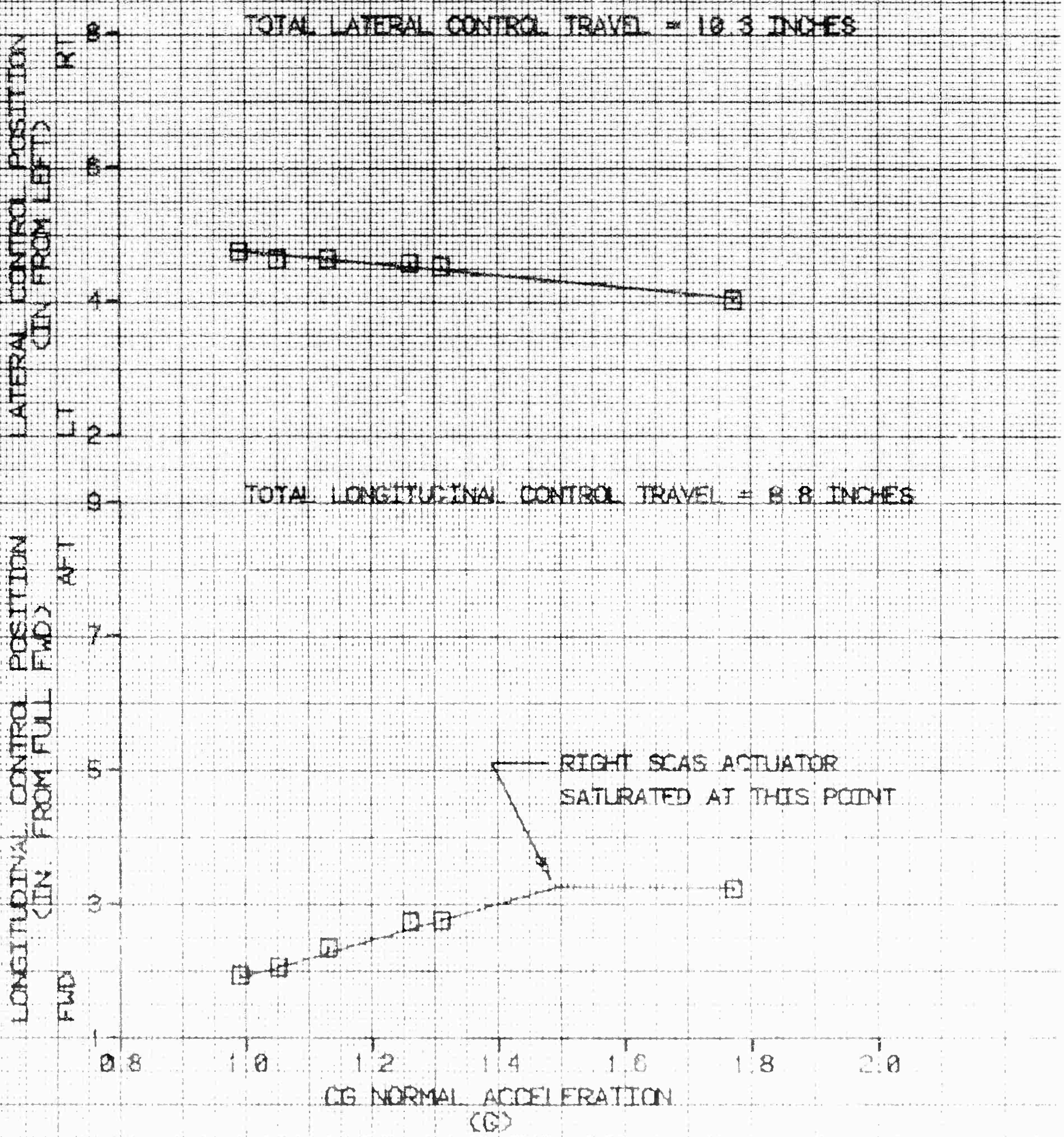


FIGURE 22A
RIGHT WIND UP TURN
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4373	111.1	9087	4.0	395	105	LEVEL

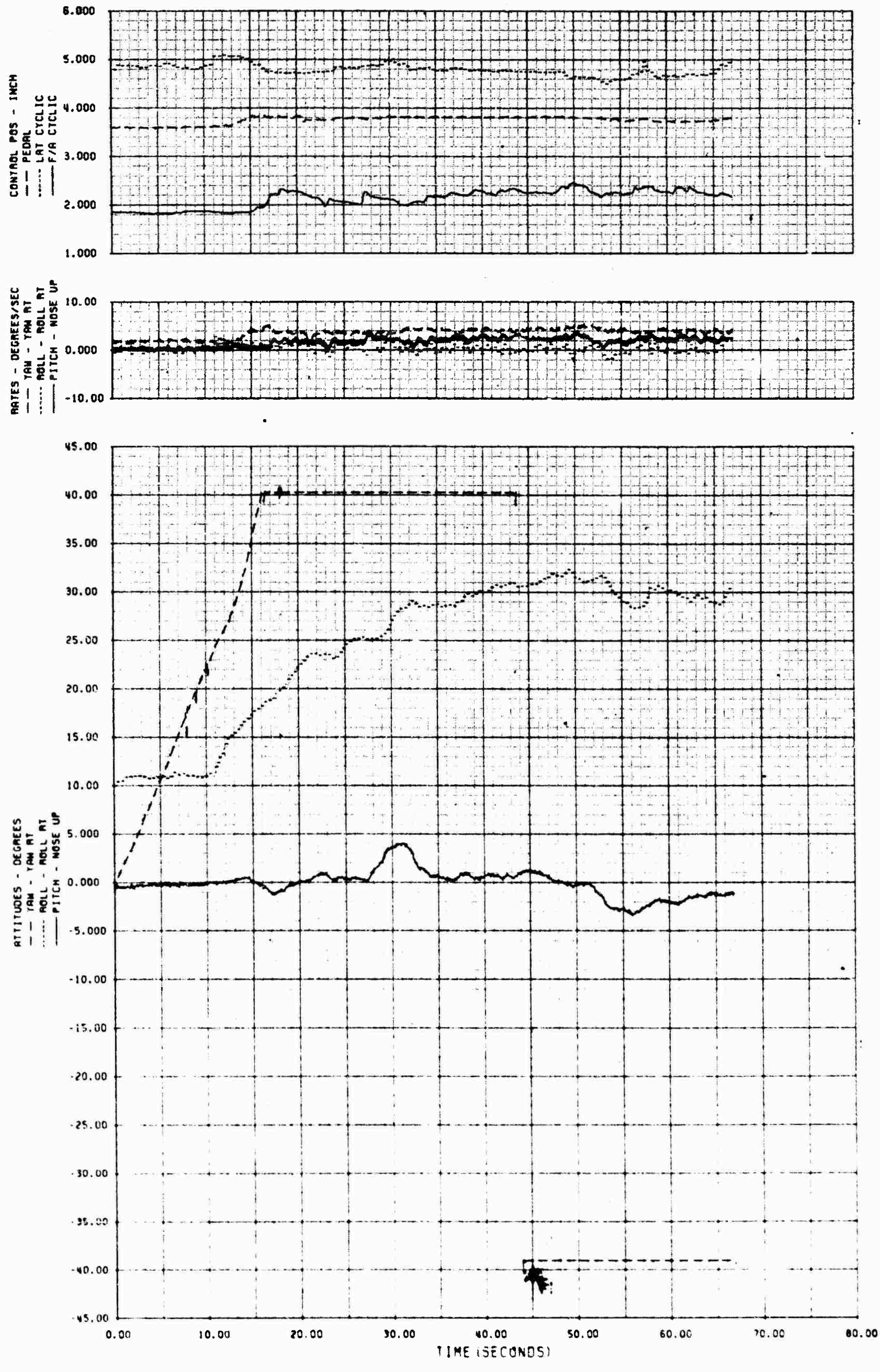


FIGURE 228
RIGHT WIND UP TURN
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4373	111.1	9087	4.0	395	105	LEVEL

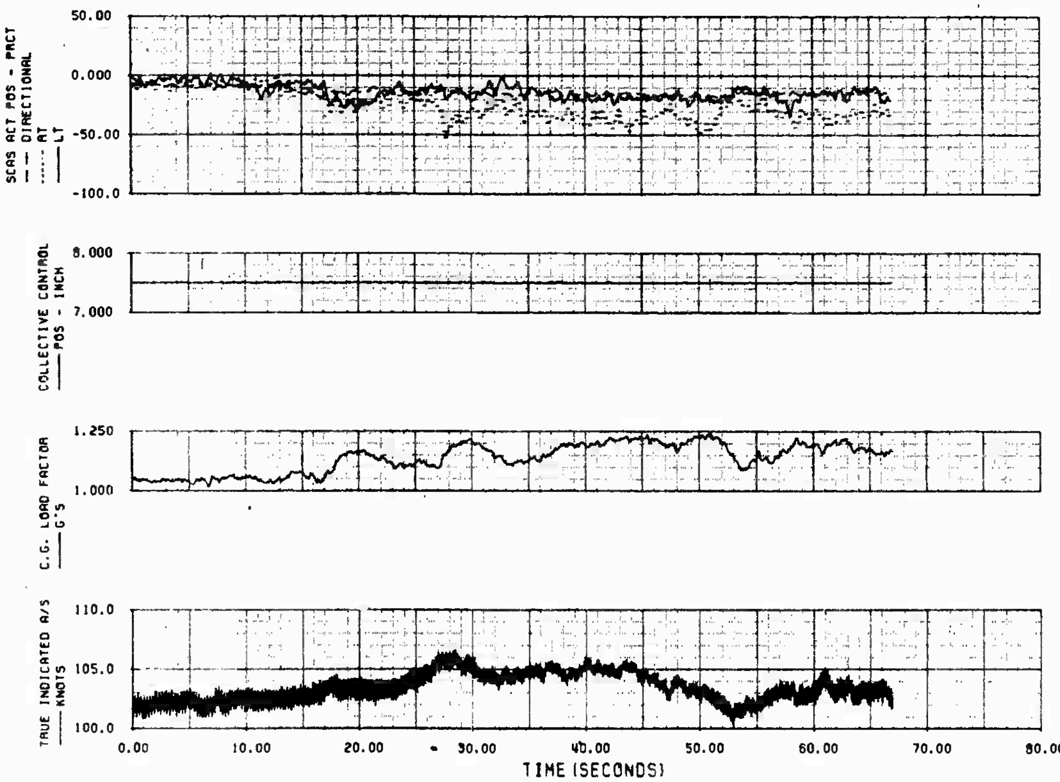
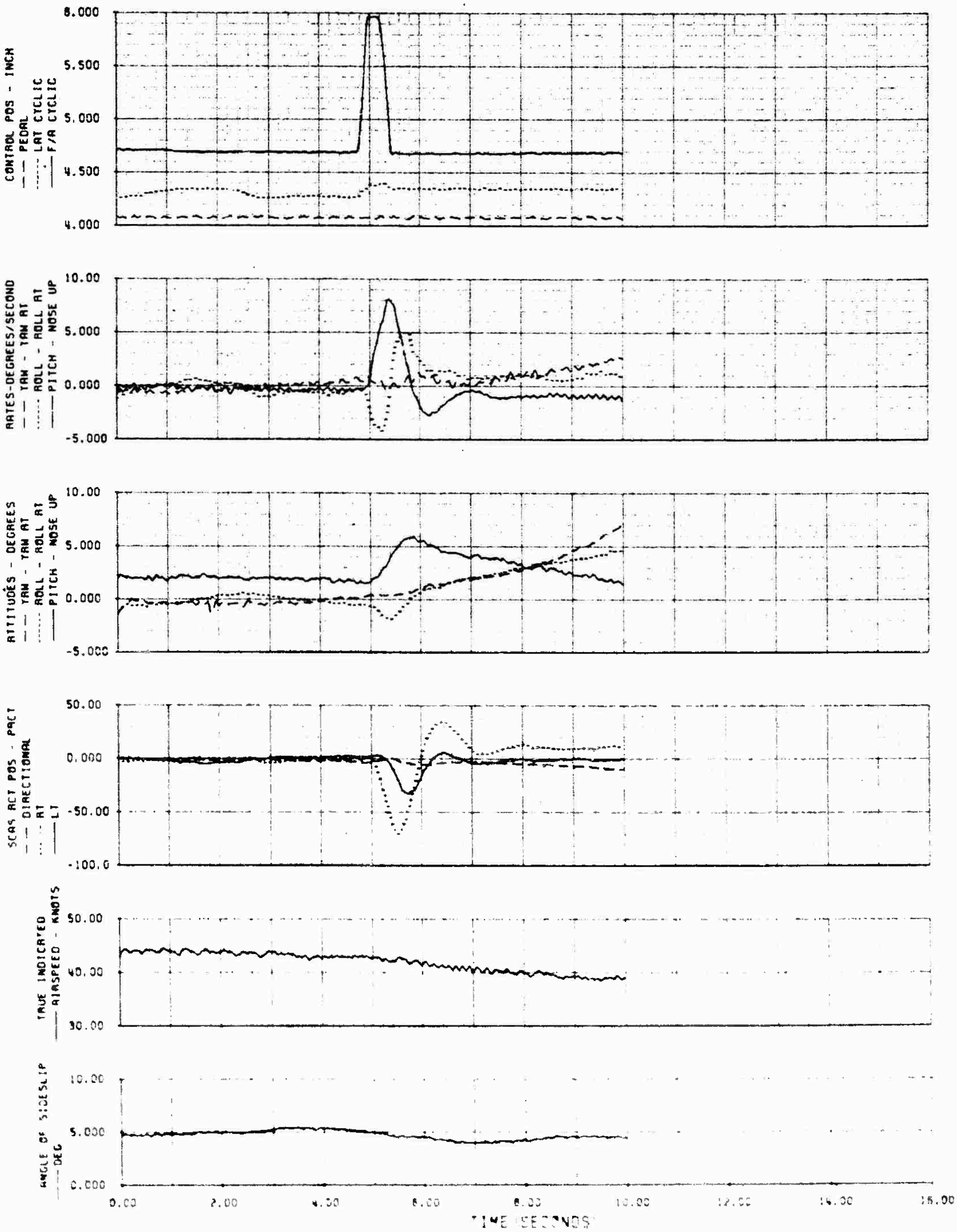


FIGURE 23
AFT LONGITUDINAL PULSE
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4393	III.ID	6132	I4.D	395	50	LEVEL



AFT LONGITUDINAL PULSE
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4393	110.80	6656	13.5	395	50	LEVEL

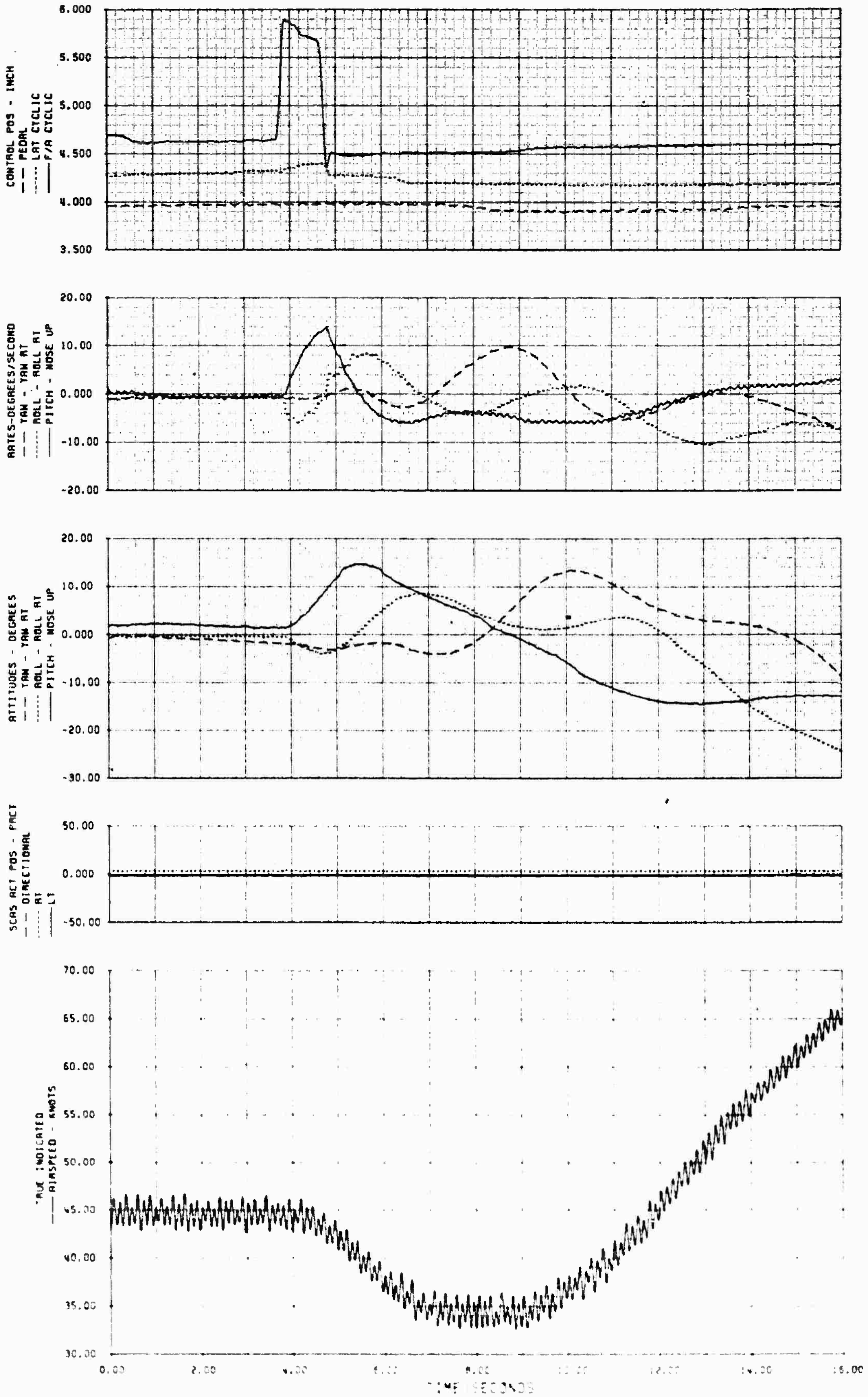


FIGURE 24B
AFT LONGITUDINAL PULSE
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4393	110.80	6656	13.5	395	50	LEVEL

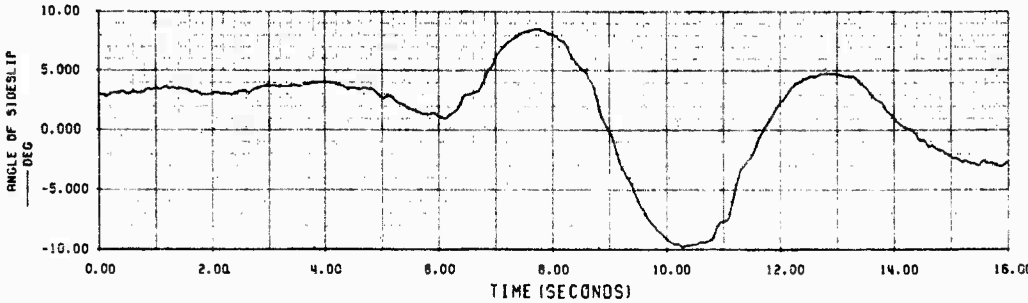


FIGURE 25A
LEFT LATERAL PULSE
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4393	110.80	6656	14.0	395	102	LEVEL

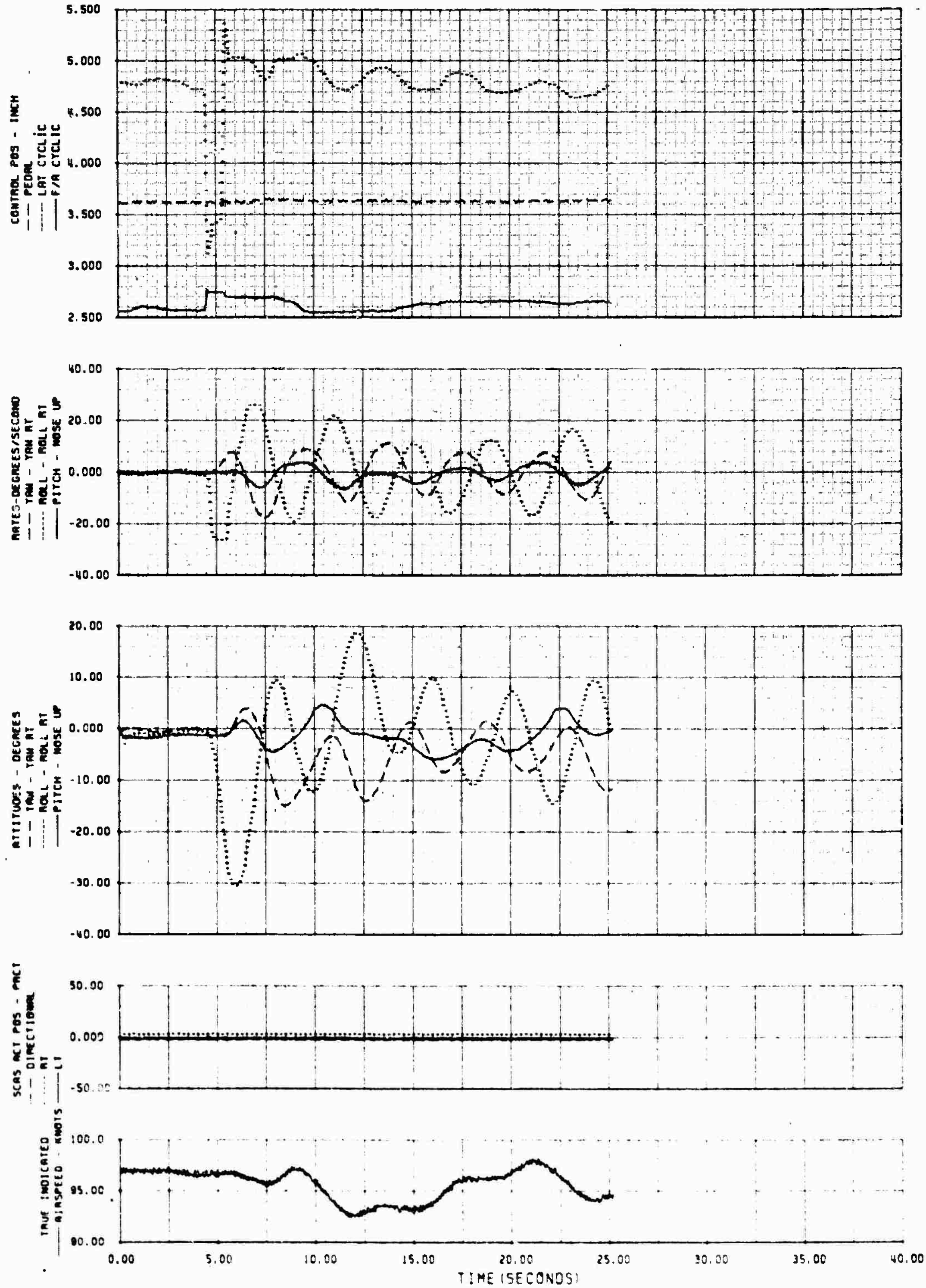


FIGURE 25B
LEFT LATERAL PULSE
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4393	110.80	6656	14.0	395	102	LEVEL

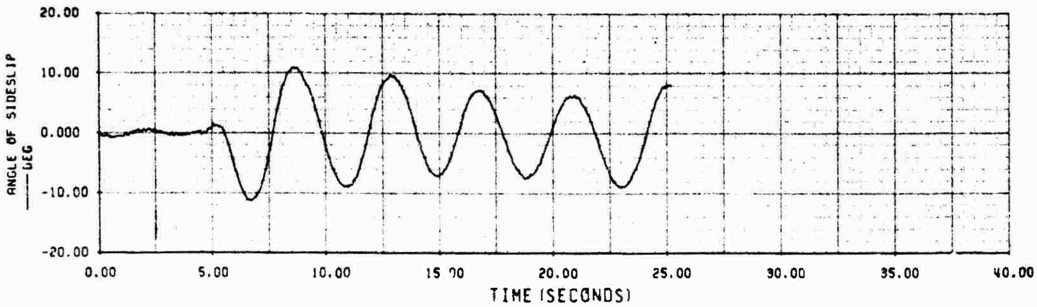


FIGURE 26A
LEFT LATERAL PULSE
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4393	111.1D	6132	14.D	395	50	LEVEL

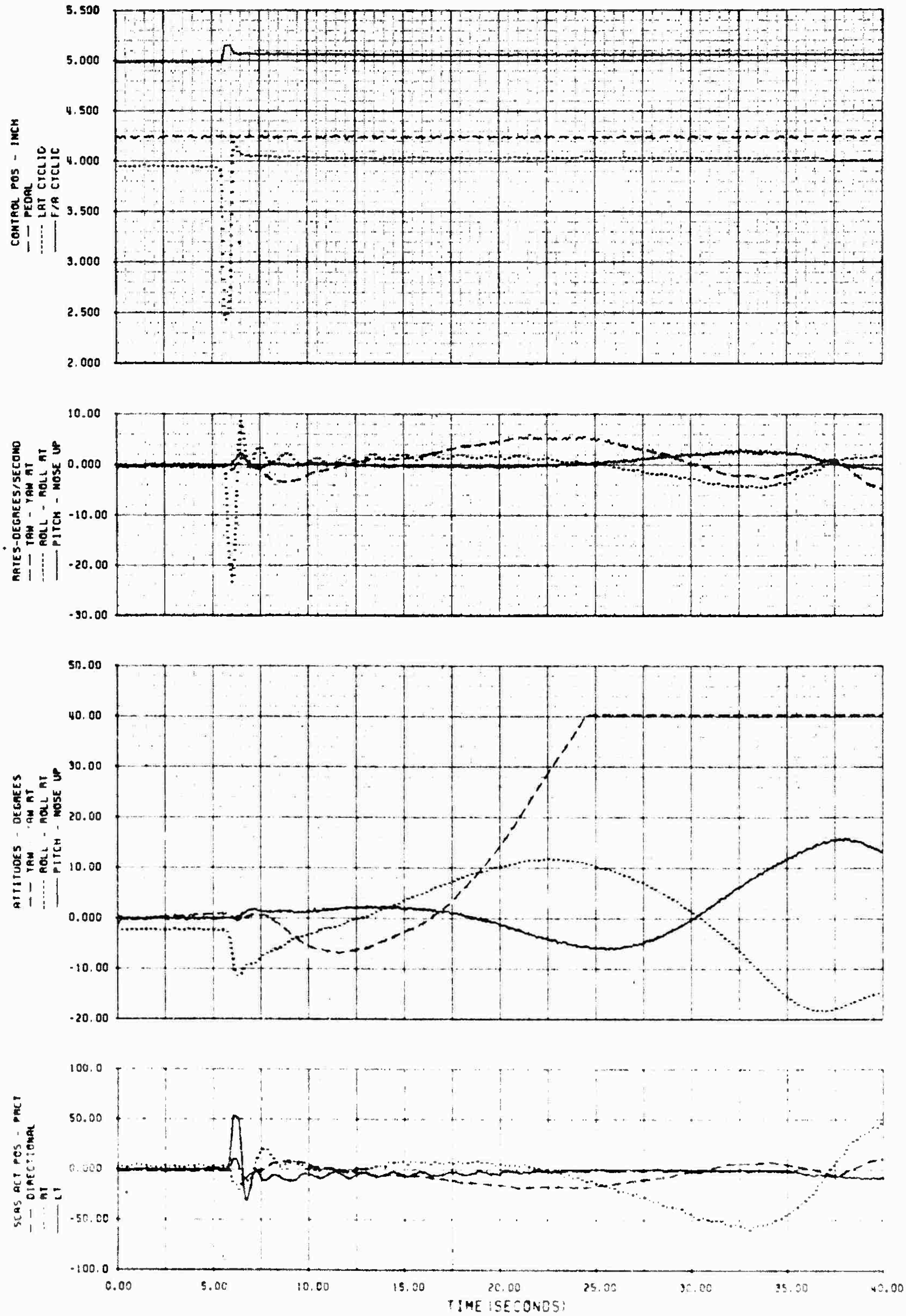


FIGURE 26B
LEFT LATERAL PULSE
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4393	111.1D	6132	14.0	395	50	LEVEL

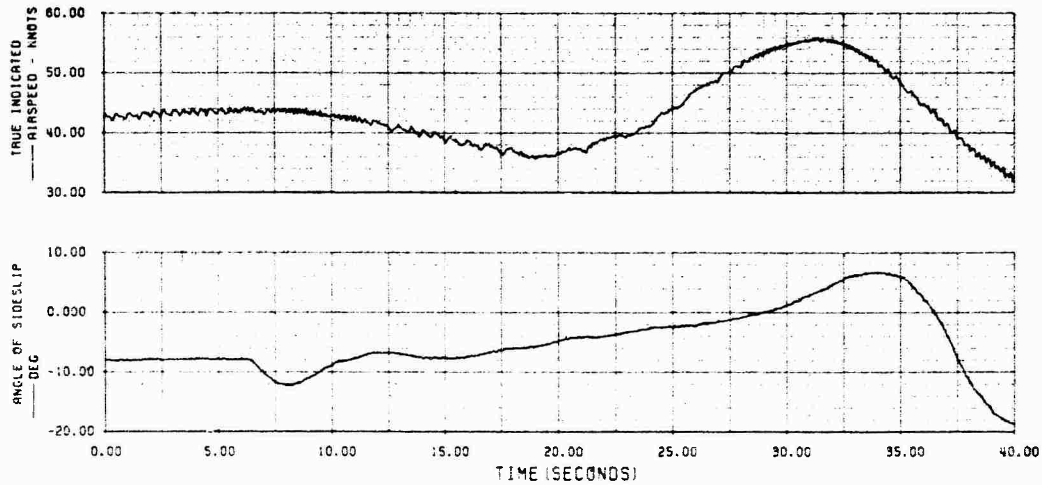


FIGURE 27A
RIGHT LATERAL PULSE AUTOROTATION AT 50 KTS (SCAS OFF)
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4378	111.30	7774	12.0	395	50	AUTOROTATION

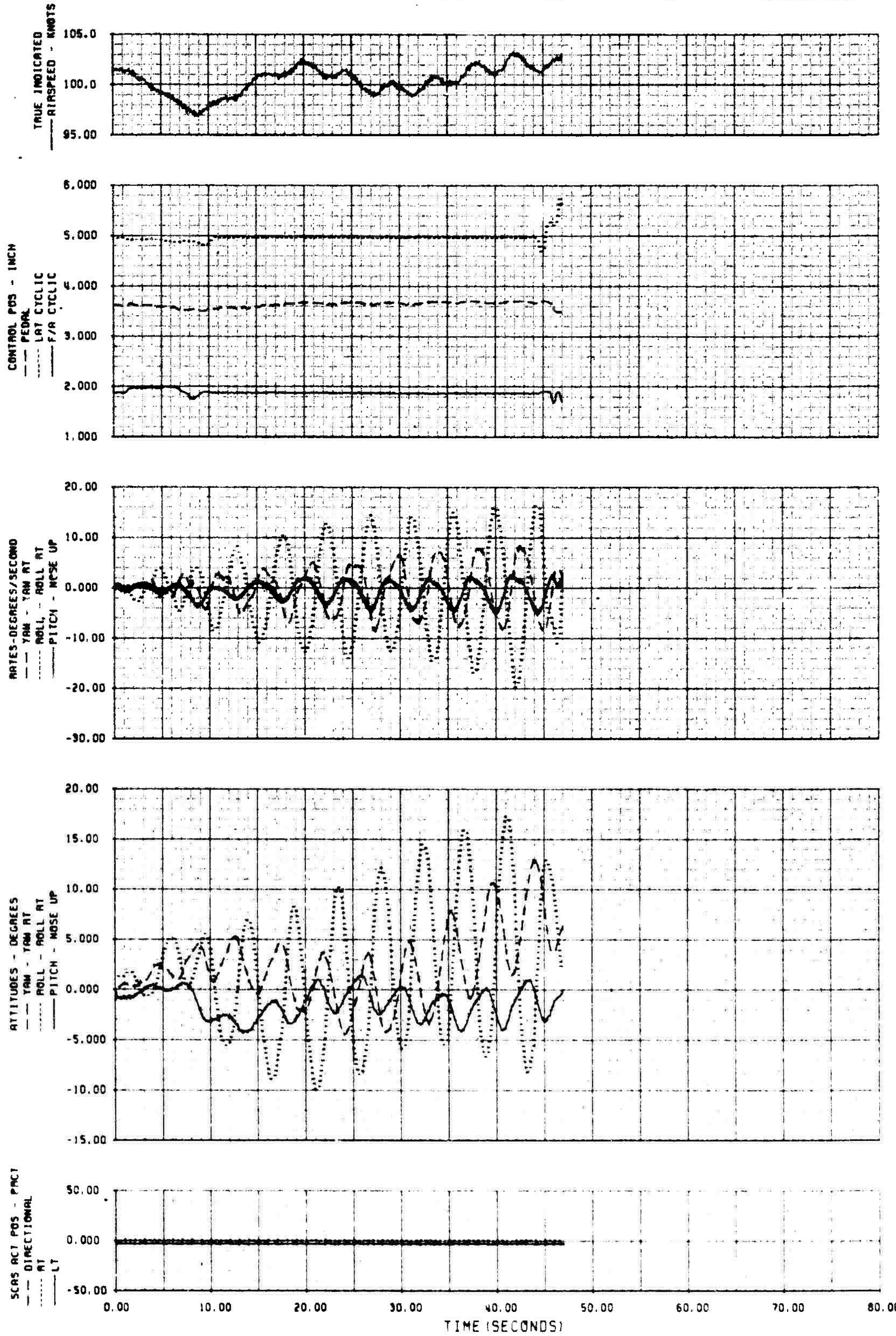


FIGURE 27B
RIGHT LATERAL PULSE AUTOROTATION AT 50 KTS (SCAS OFF)
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4378	111.30	7774	12.0	395	50	AUTOROTATION

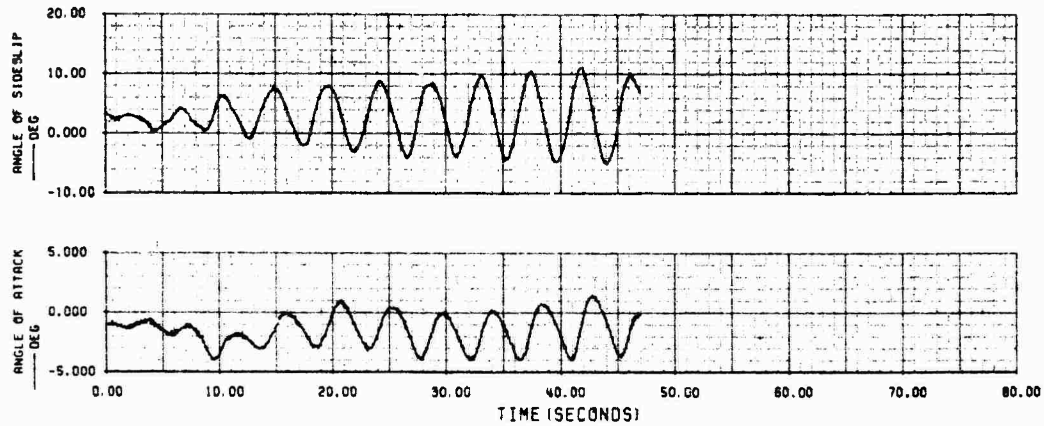


FIGURE 28
LONG PERIOD - LEVEL FLIGHT AT 102 KTS
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4378	111.30	7593	12.5	395	102	LEVEL

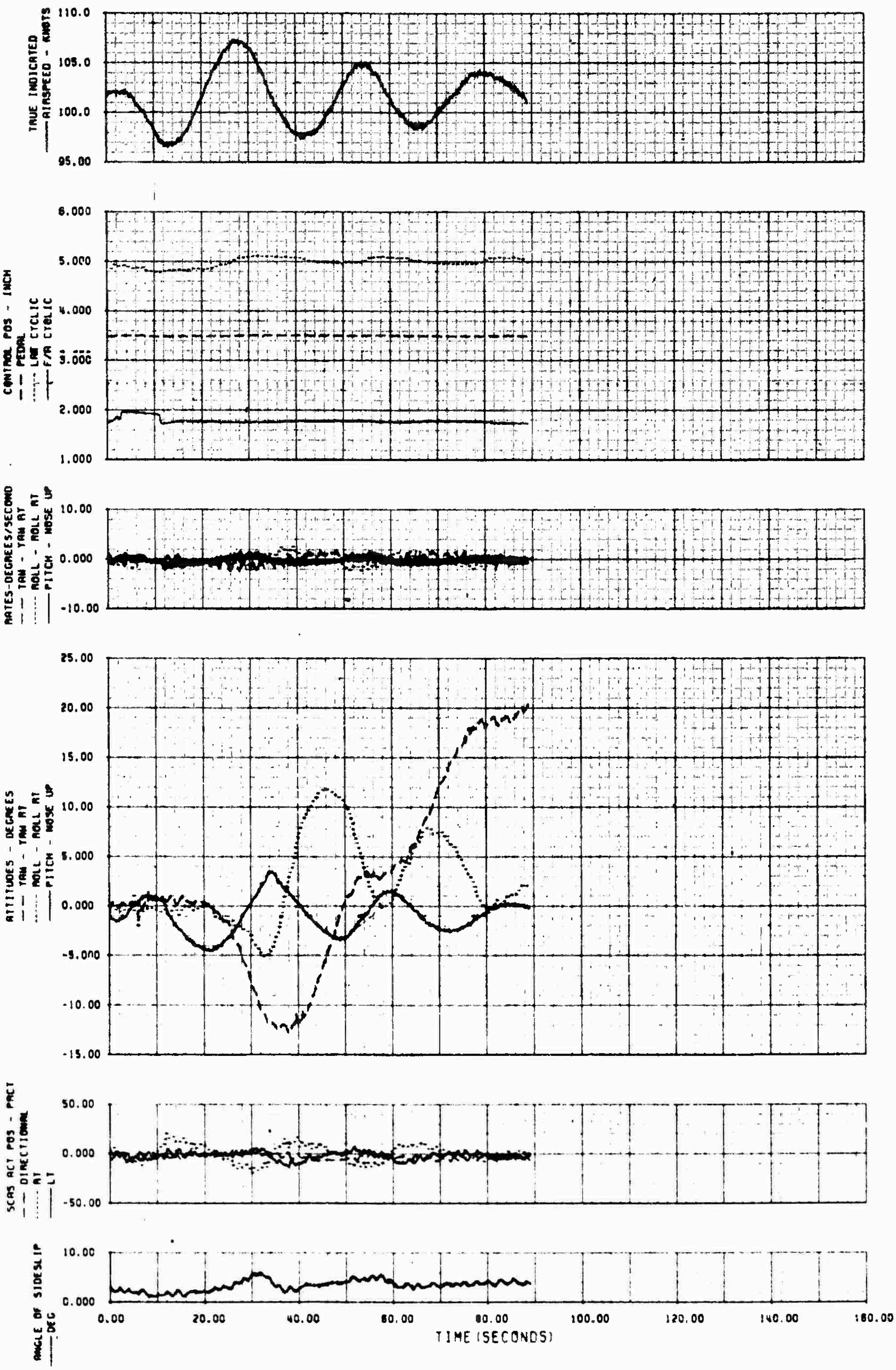


FIGURE 29A
LONG PERIOD - LEVEL FLIGHT AT 50 KTS (SCAS ON)
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4378	111.30	7867	12.5	395	50	LEVEL

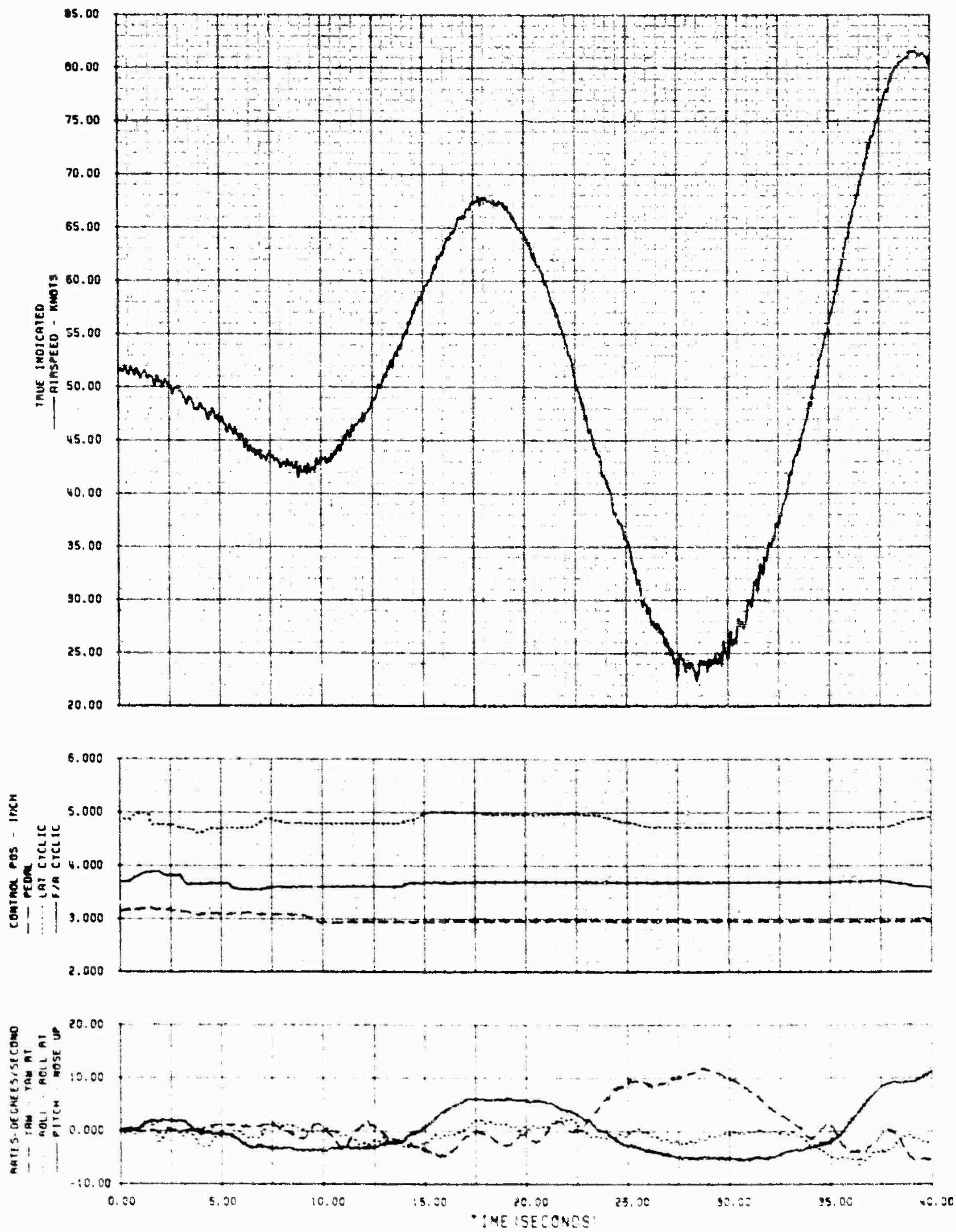


FIGURE 29B
LONG PERIOD - LEVEL FLIGHT AT 50 KTS (SCAS ON)
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4378	111.30	7867	12.5	395	50	LEVEL

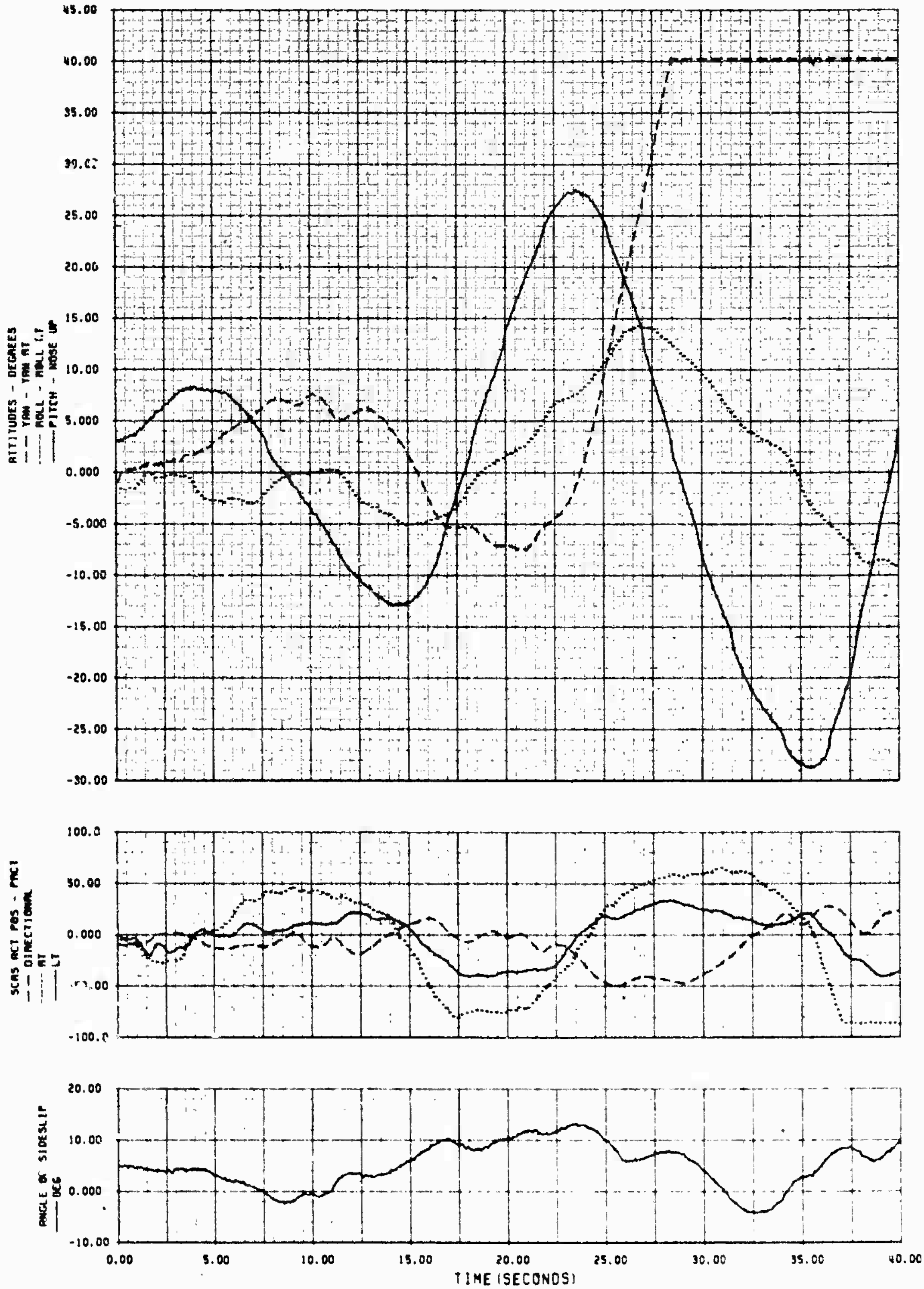


FIGURE 29C
LONG PERIOD - LEVEL FLIGHT AT 50 KTS (SCAS ON)
OH-580 USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4378	111.30	7867	12.5	395	50	LEVEL

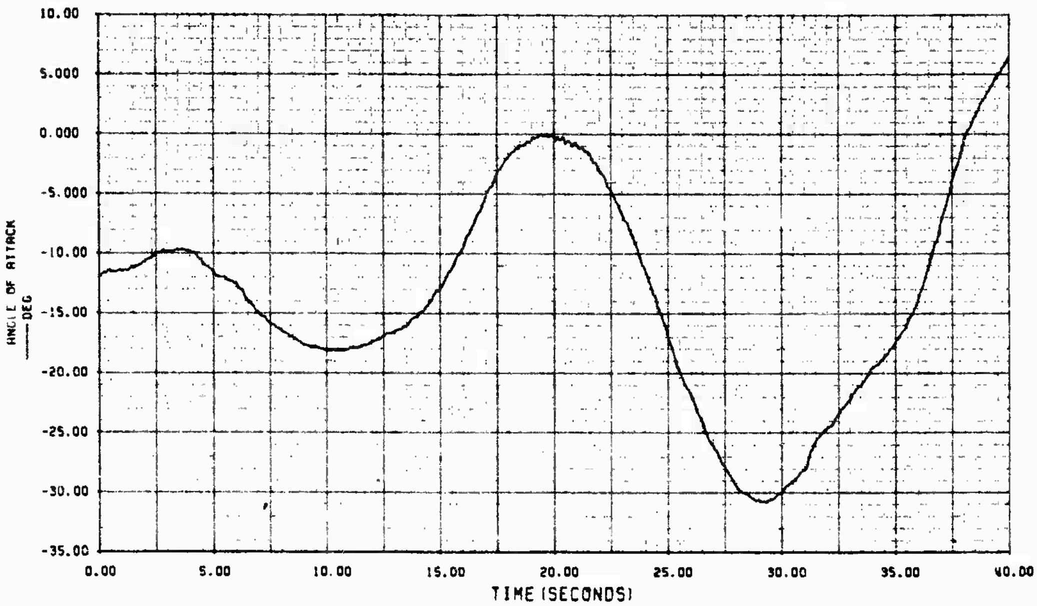
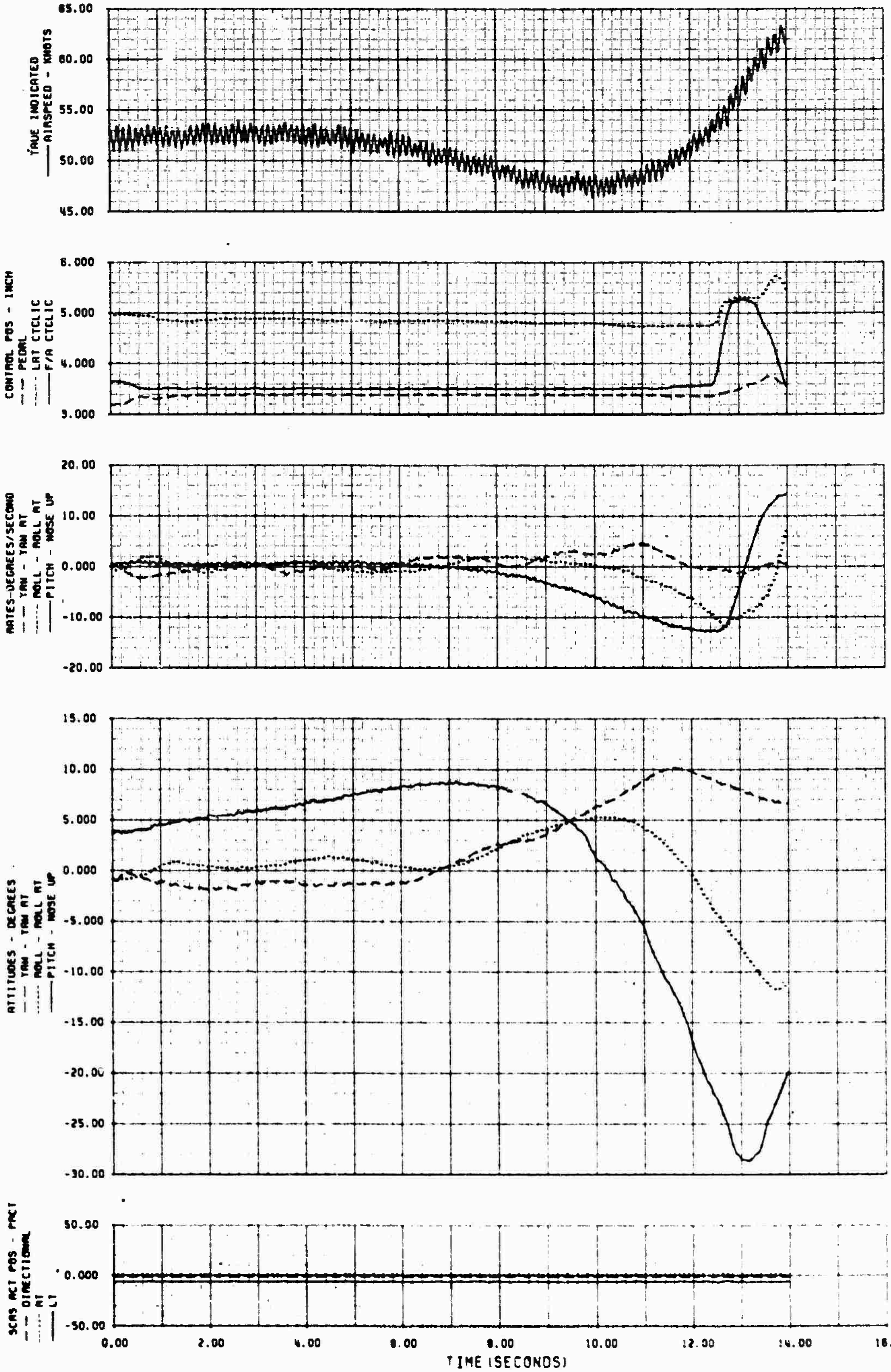


FIGURE 30A
LONG PERIOD CLIMB AT 50 KTS (SCAS OFF)
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4378	111.3D	9050	8.D	395	56	CLIMB



LONG PERIOD-CLIMB AT 50 KTS (SCAS OFF)

FIGURE 30B
LONG PERIOD CLIMB AT 50 KTS (SCAS OFF)
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4378	111.30	9050	8.0	395	56	CLIMB

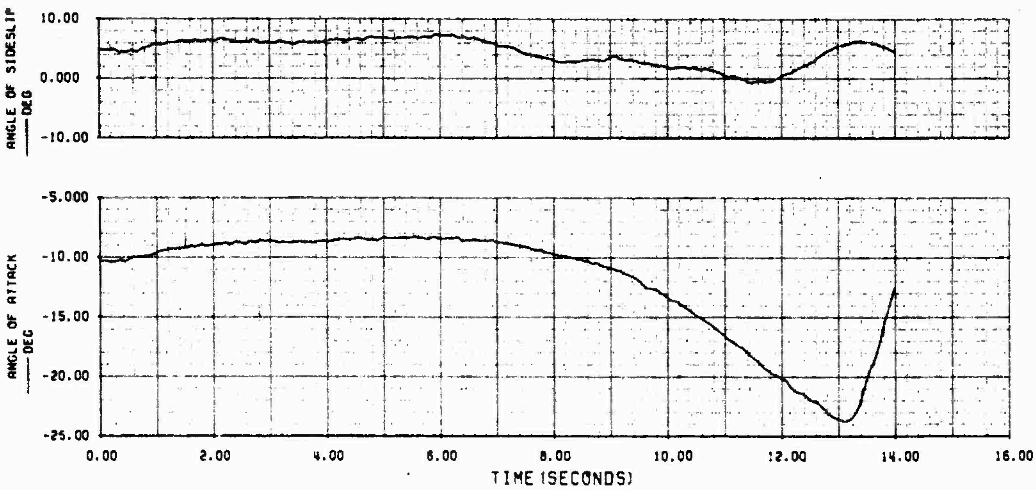


FIGURE 31
DIRECTIONAL CONTROLLABILITY
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
3900	112.3 (AFT)	1980	22.0	395	0	HOVER

NOTES: 1. SCAS ON
2. MAST MOUNTED SIGHT REMOVED
3. SMALL STABILIZER INSTALLED

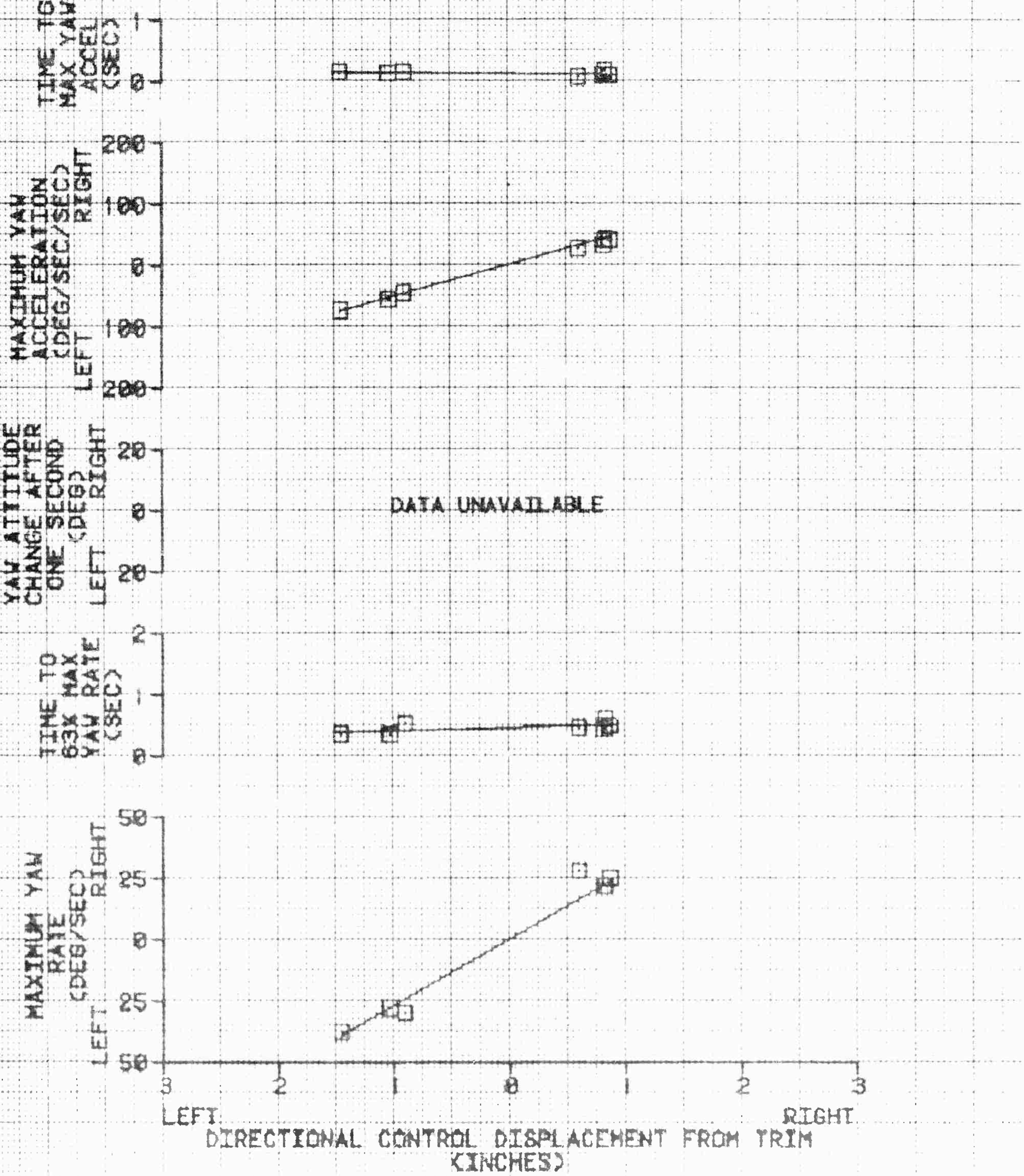


FIGURE 32
DIRECTIONAL CONTROLLABILITY
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4210	111.0 (AFT)	1390	20.0	395	0	HOVER

NOTES: 1. SCAS ON
2. SMALL STABILIZER INSTALLED

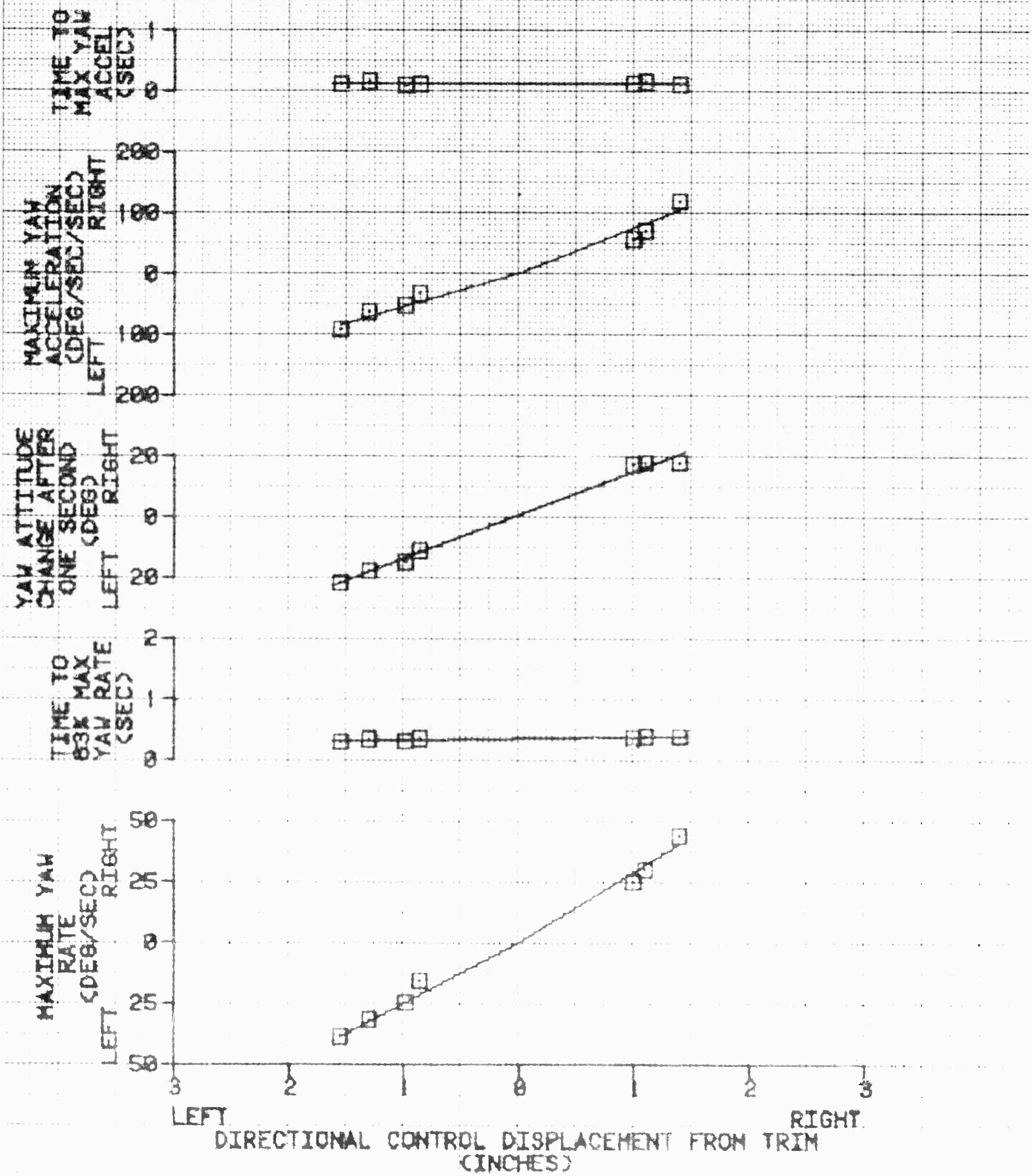


FIGURE 33

DIRECTIONAL CONTROLLABILITY
OH-58D USA S/N 89-18285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4500	110.4 (AFT)	2330	27.0	395	0	HOVER

NOTES: 1. SCAS ON
2. SMALL STABILIZER INSTALLED

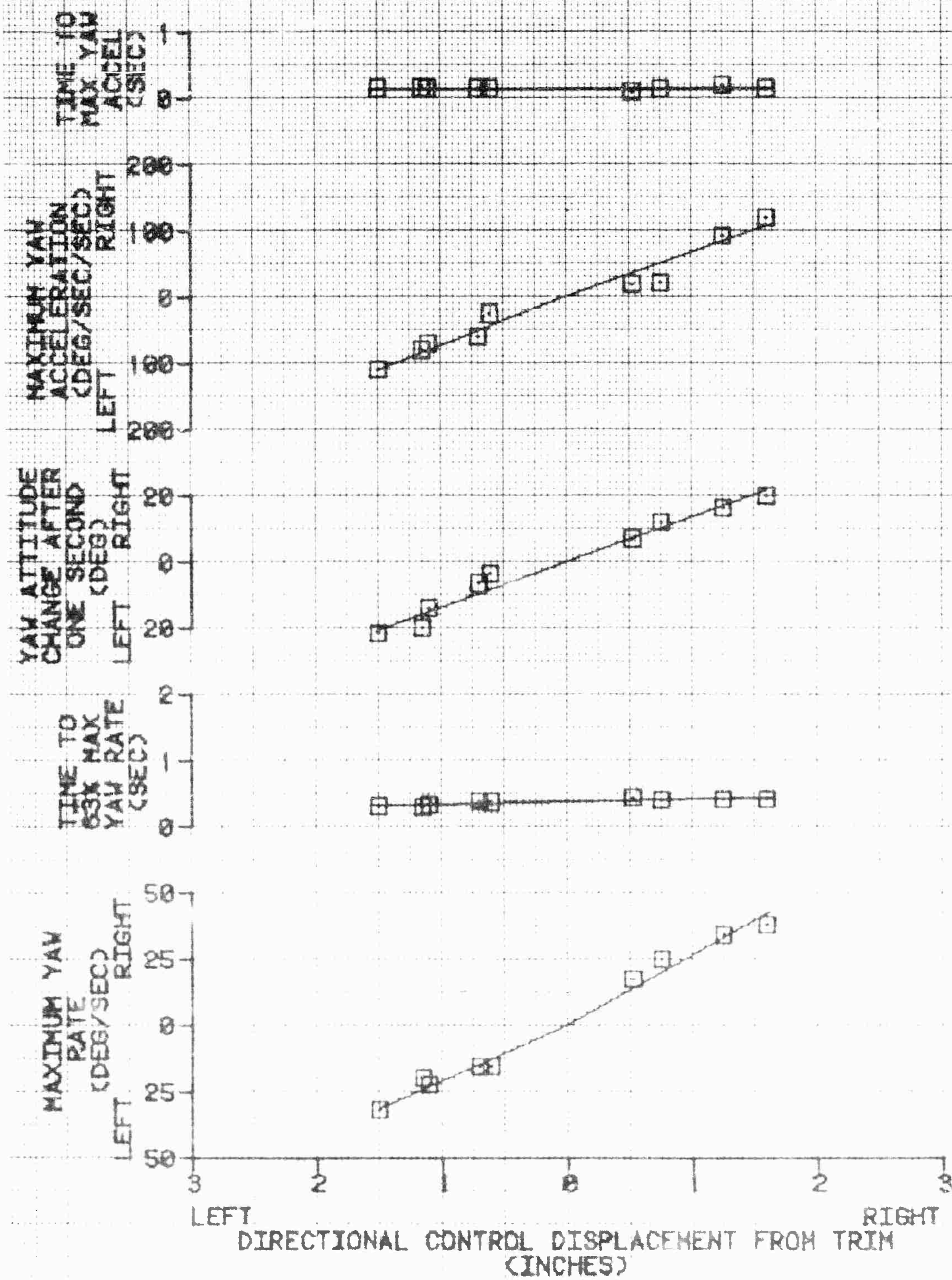


FIGURE 34
LONGITUDINAL CONTROLLABILITY
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
3040	112.8 (AFT)	1980	22.0	396	0	HOVER

NOTES: 1. SCAS ON
2. MAST MOUNTED SIGHT REMOVED
3. SMALL STABILIZER INSTALLED

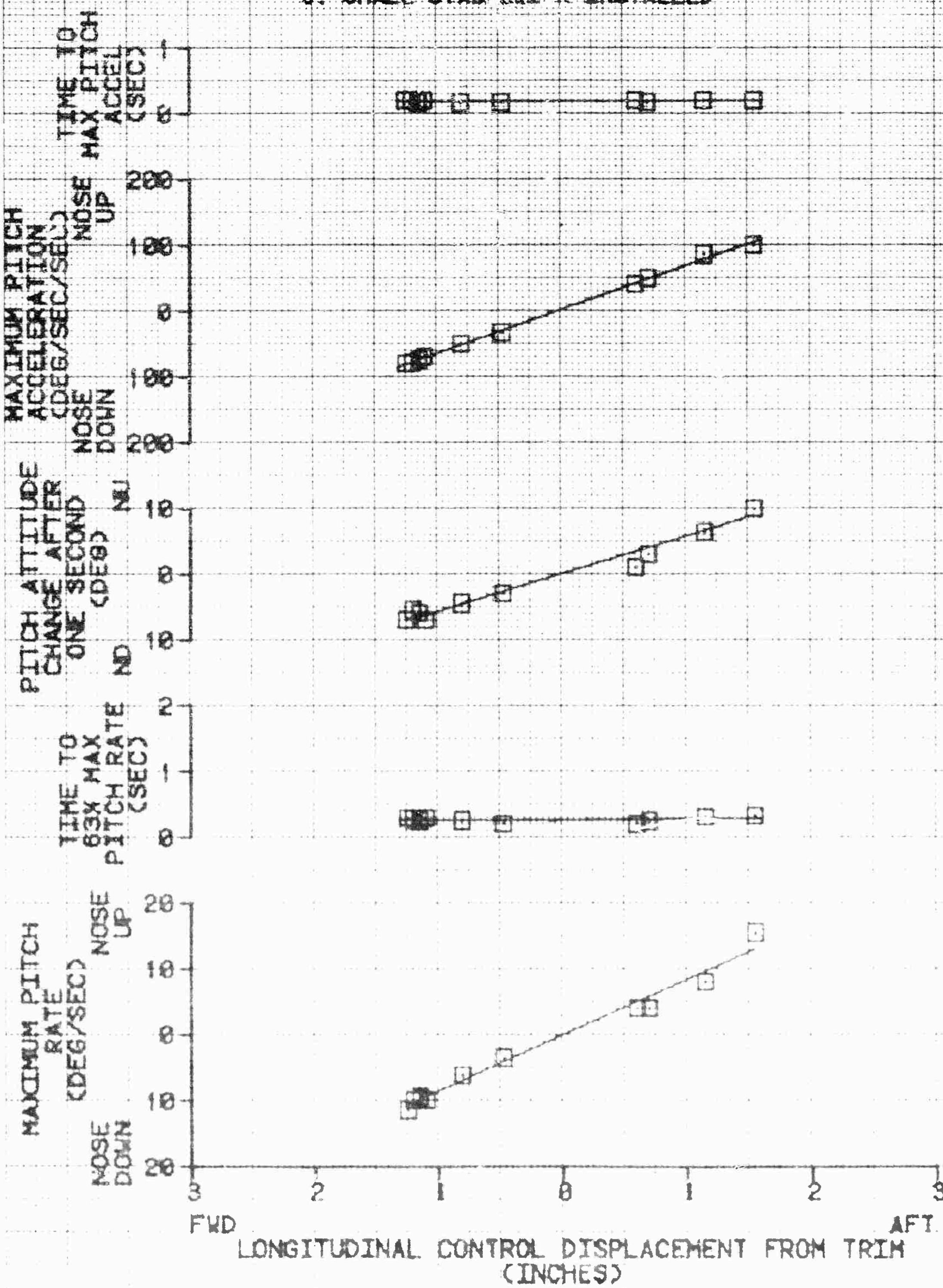


FIGURE 35
LONGITUDINAL CONTROLLABILITY
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
3950	112.1 (AFT)	8540	17.0	395	102	LEVEL

NOTES: 1. SCAS ON
2. MAST MOUNTED SIGHT REMOVED
3. SMALL STABILIZER INSTALLED

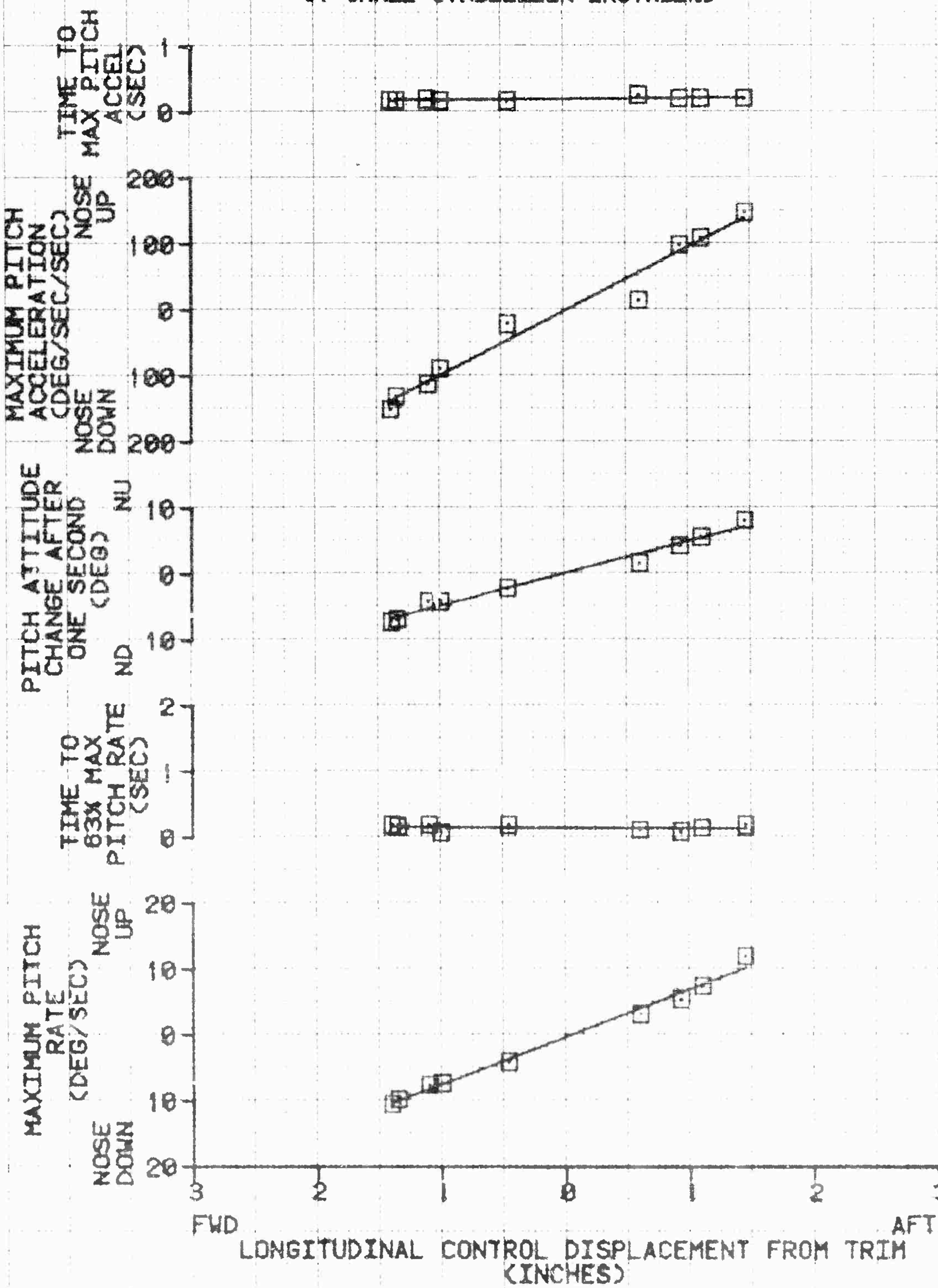


FIGURE 36

LONGITUDINAL CONTROLLABILITY
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4210	111.0 (AFT)	1330	28.0	395	0	HOVER

NOTES: 1. SCAS ON
2. SMALL STABILIZER INSTALLED

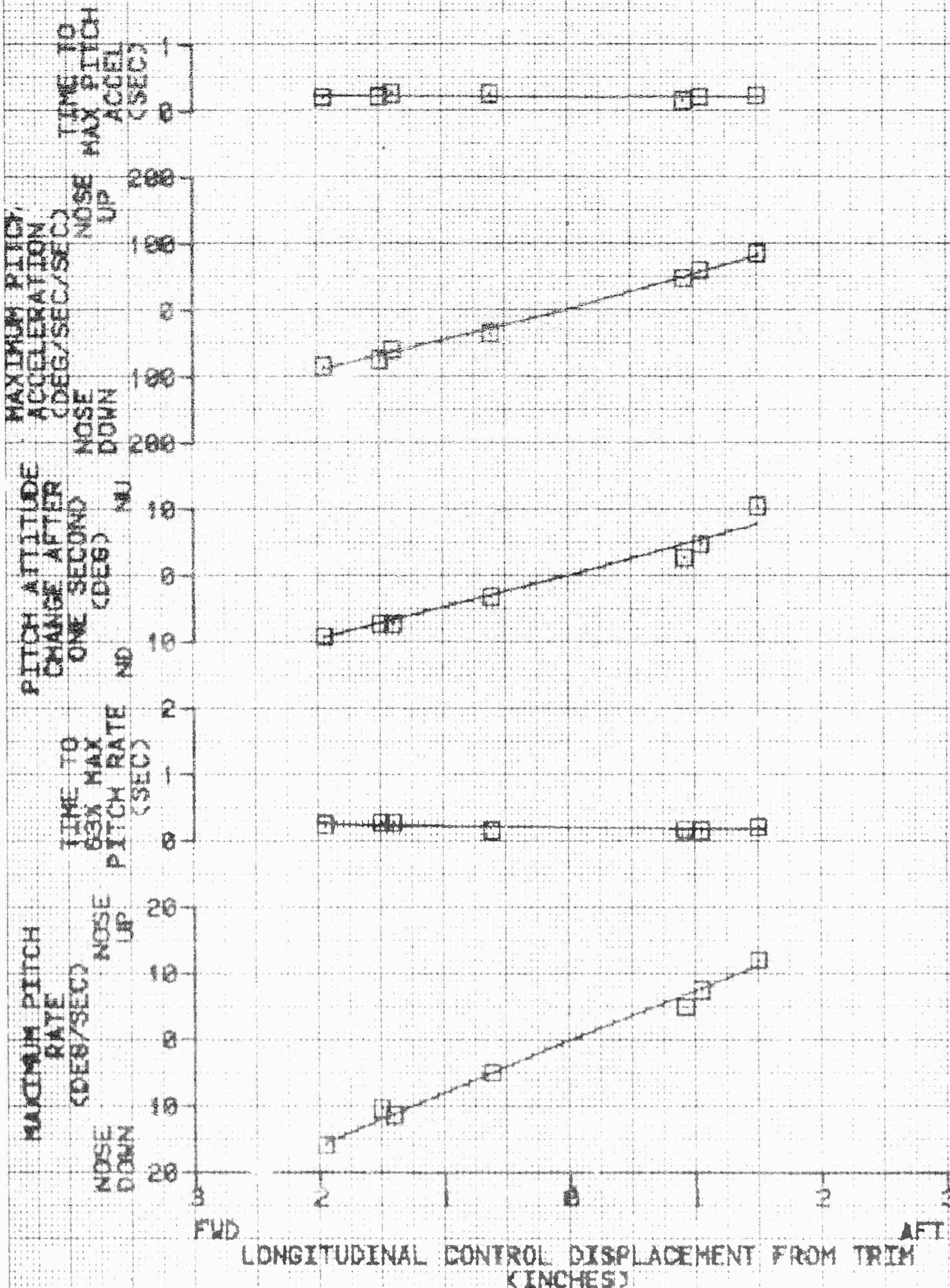


FIGURE 37

LONGITUDINAL CONTROLLABILITY
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4210	111.0 (AFT)	8500	10.5	395	102	LEVEL

NOTES: 1. SCAS ON
2. SMALL STABILIZER INSTALLED

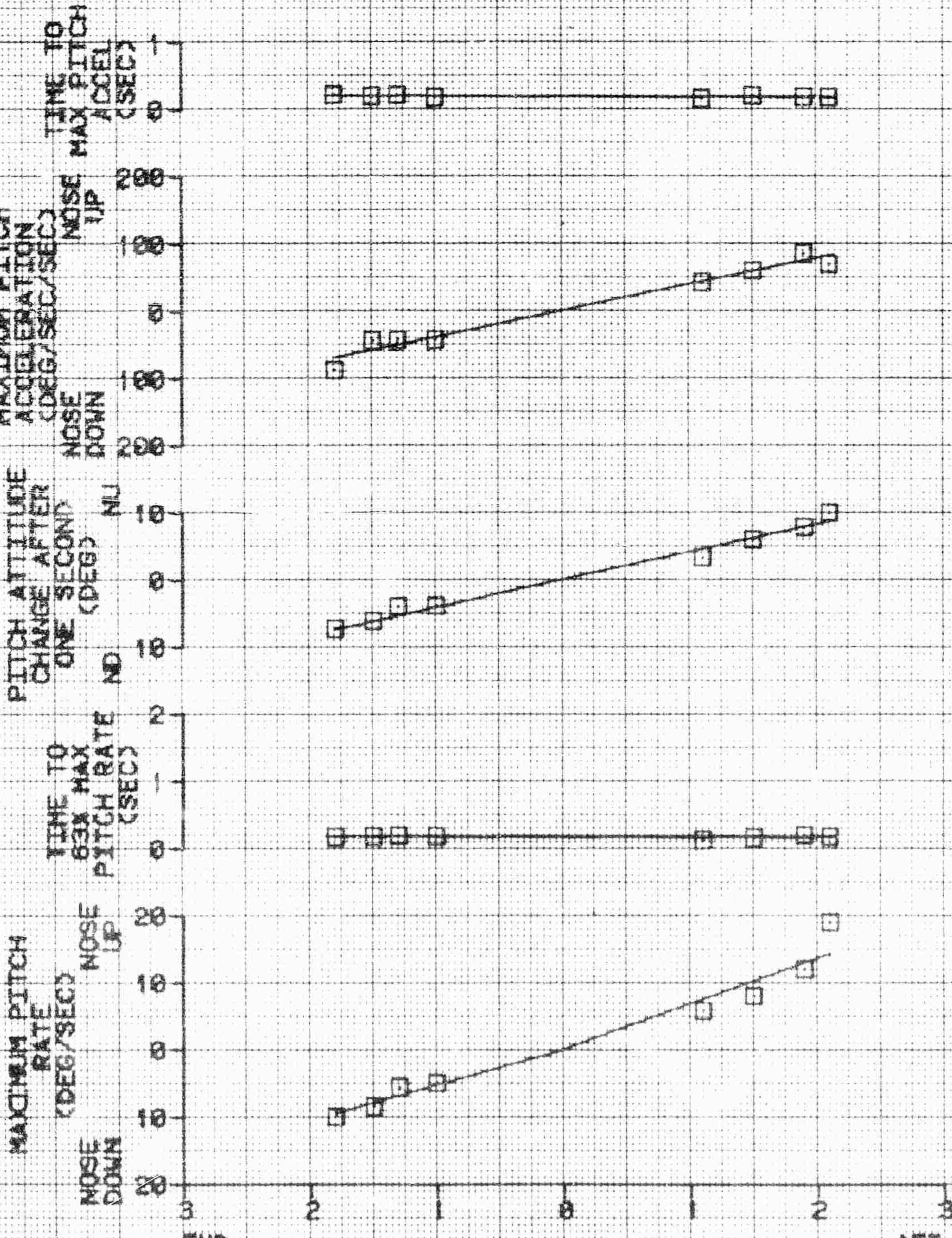


FIGURE 38

LONGITUDINAL CONTROLLABILITY

OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4500	110.4 (AFT)	2300	27.0	395	0	HOVER

NOTES: 1. SCAS ON
2. SMALL STABILIZER INSTALLED

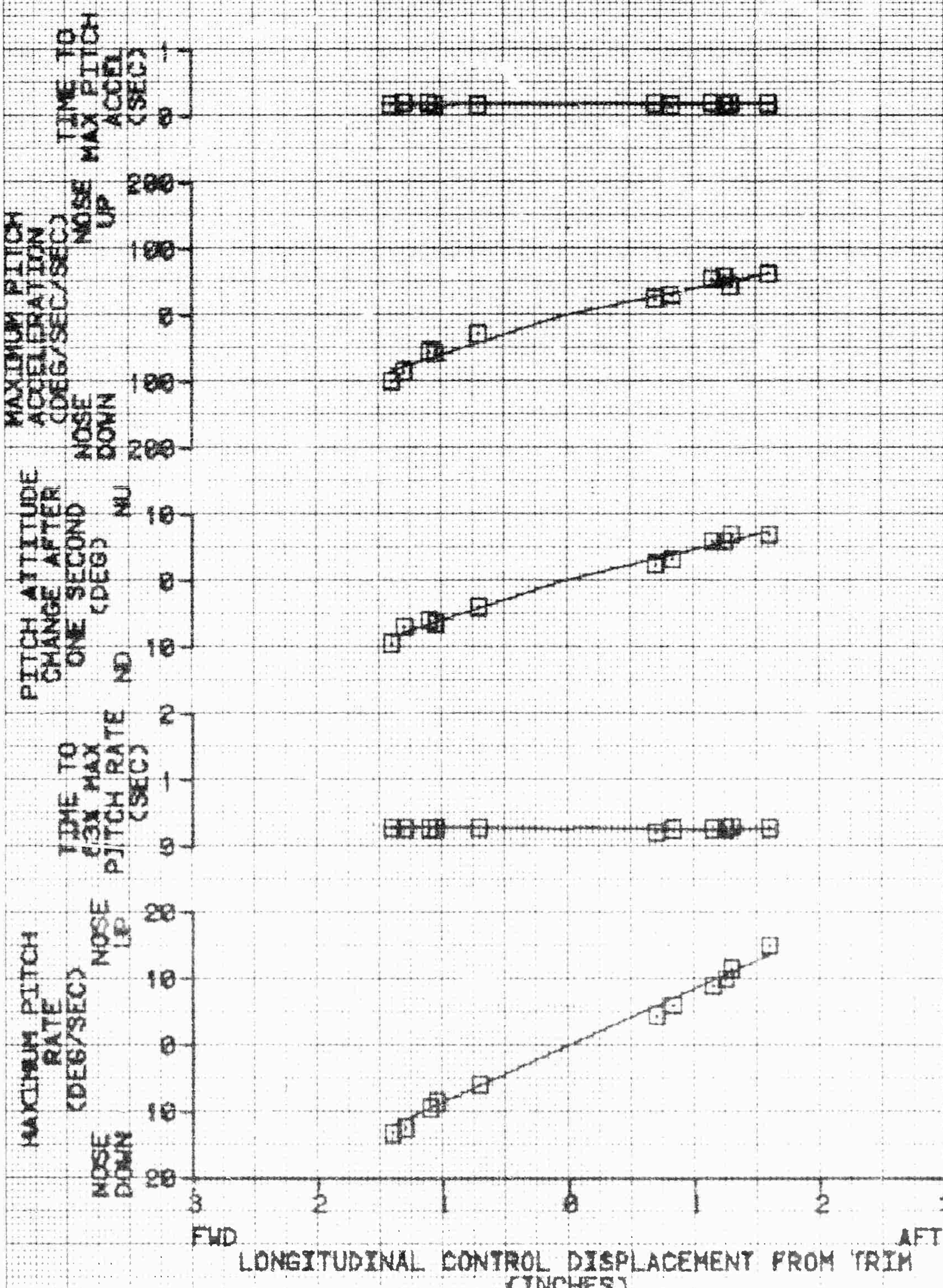


FIGURE 89

LONGITUDINAL CONTROLLABILITY
OH-58D USA S/N 89-18285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4480	110.3 (AFT)	6920	18.0	395	102	LEVEL

NOTES: 1. SCAS ON
2. SMALL STABILIZER INSTALLED

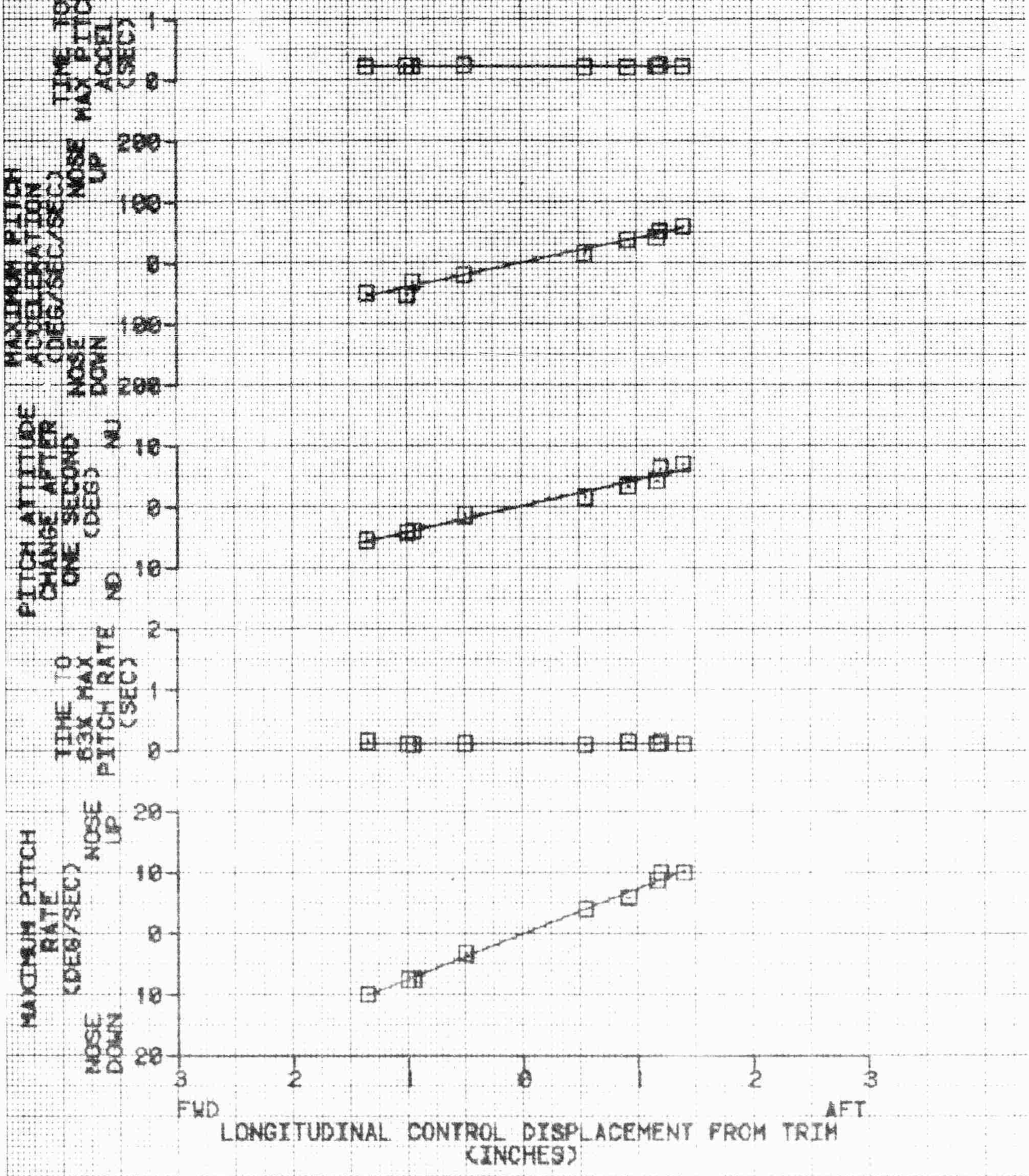


FIGURE 40
LATERAL CONTROLLABILITY
OH-58D USA S/N 69-16285

SYM	AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
⊡	4210	111.0 (AFT)	1930	20.0	395	0	HOVER
△	4240	110.6 (AFT)	2179	28.0	395	0	HOVER

NOTES: 1. SCAS ON
2. ⊡ DENOTES SMALL STABILIZER AND OLD ROLL SCAS GAINS
3. △ DENOTES EXTENDED STABILIZER AND NEW ROLL SCAS GAINS

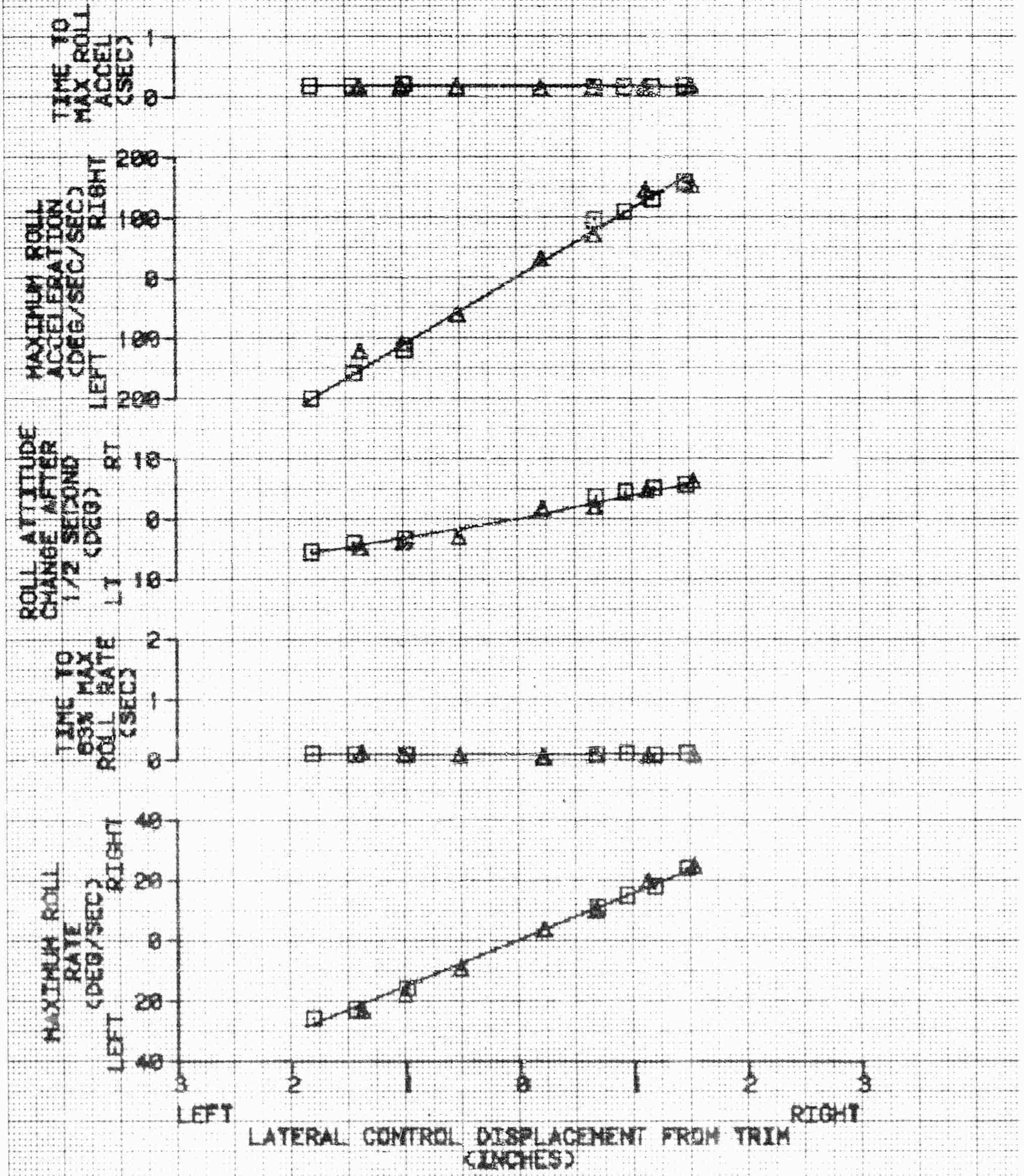


FIGURE 41
LATERAL CONTROLLABILITY
OH-58D USA S/N 69-16285

SYM	AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
□	3050	112.1 (AFT)	6540	17.0	395	102	LEVEL
△	3090	111.2 (AFT)	[illegible]	15.6	395	102	LEVEL

NOTES:
1. SCAS ON
2. MAST MOUNTED SIGHT REMOVED
3. □ DENOTES SMALL STABILIZER AND OLD ROLL SCAS GAINS
4. △ DENOTES EXTENDED STABILIZER AND NEW ROLL SCAS GAINS

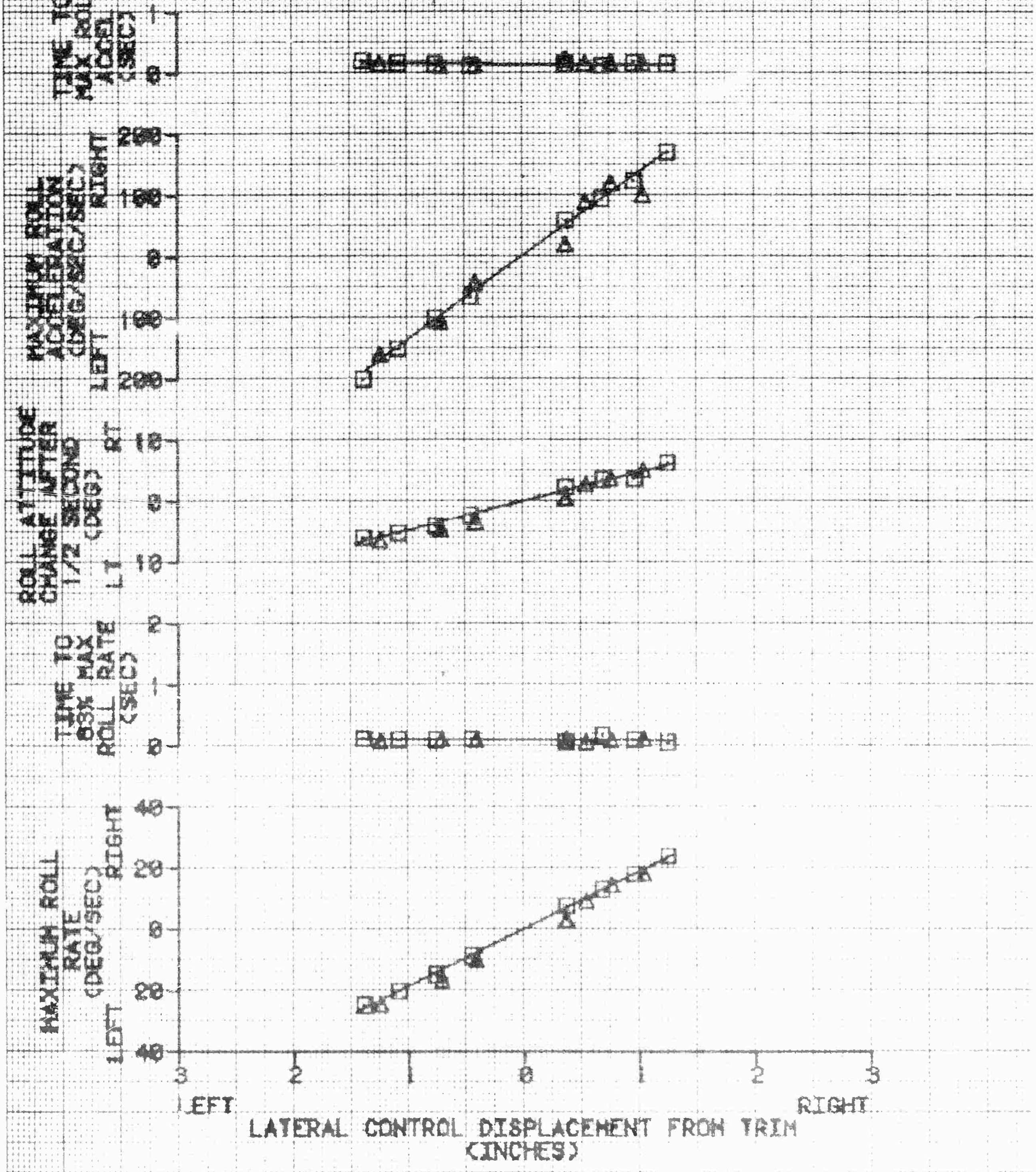


FIGURE 42

LATERAL CONTROLLABILITY

OH-58D USA S/N 69-16285

SYM	AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
□	4250	111.0 (AFT)	6400	17.5	395	102	LEVEL
△	4230	110.5 (AFT)	5880	18.0	395	102	LEVEL

NOTES:
1. SCAS ON
2. □ DENOTES SMALL STABILIZER AND OLD ROLL SCAS GAINS
3. △ DENOTES EXTENDED STABILIZER AND NEW ROLL SCAS GAINS

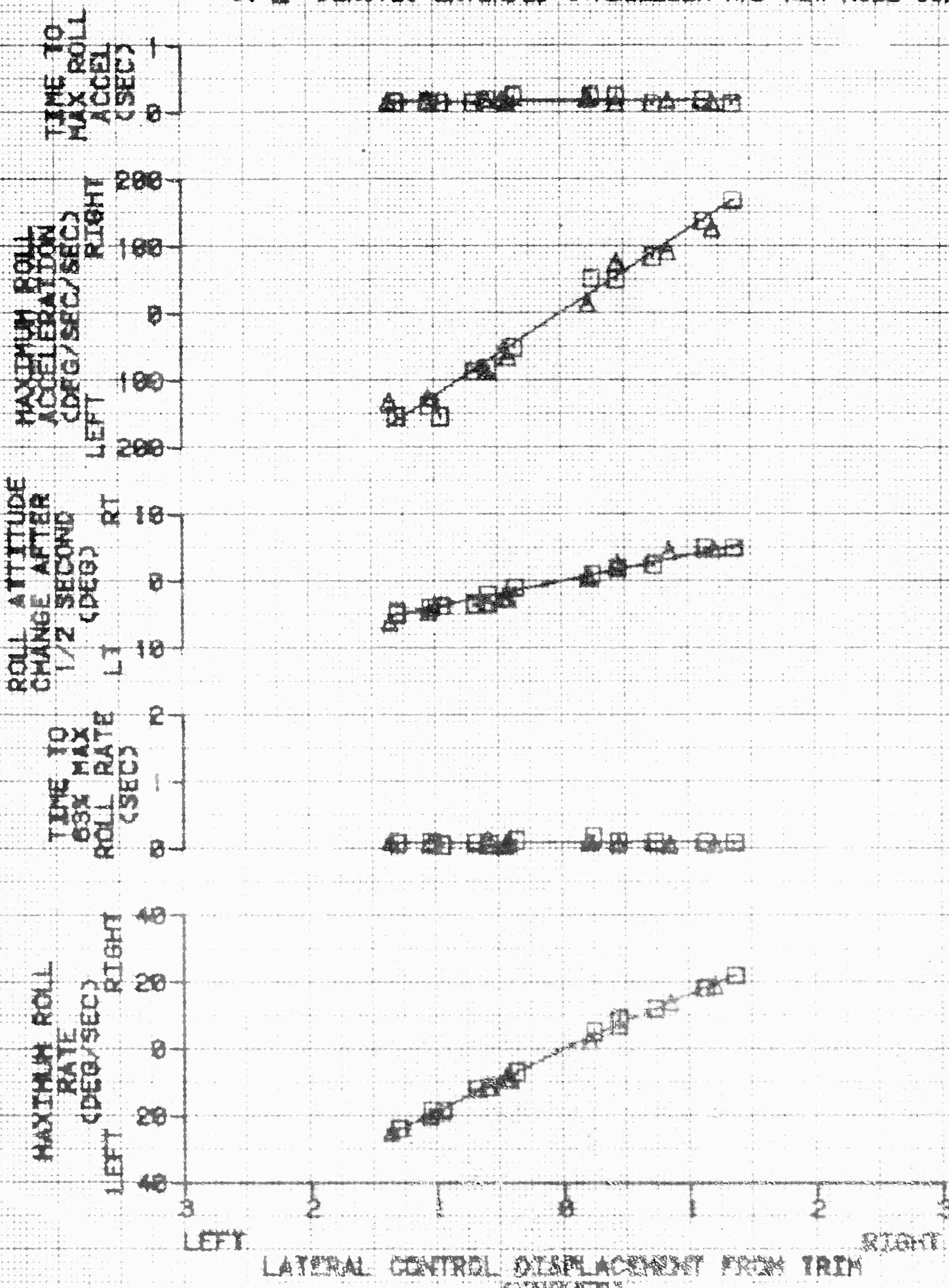


FIGURE 43

ROLL RESPONSE COMPARISON
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	SCAS GAIN
4610	110.4(AFT)	680	15.0	395	OLD
4350	111.0(AFT)	840	22.0	395	NEW

NOTE: 0.9 INCH LEFT STEP INPUT

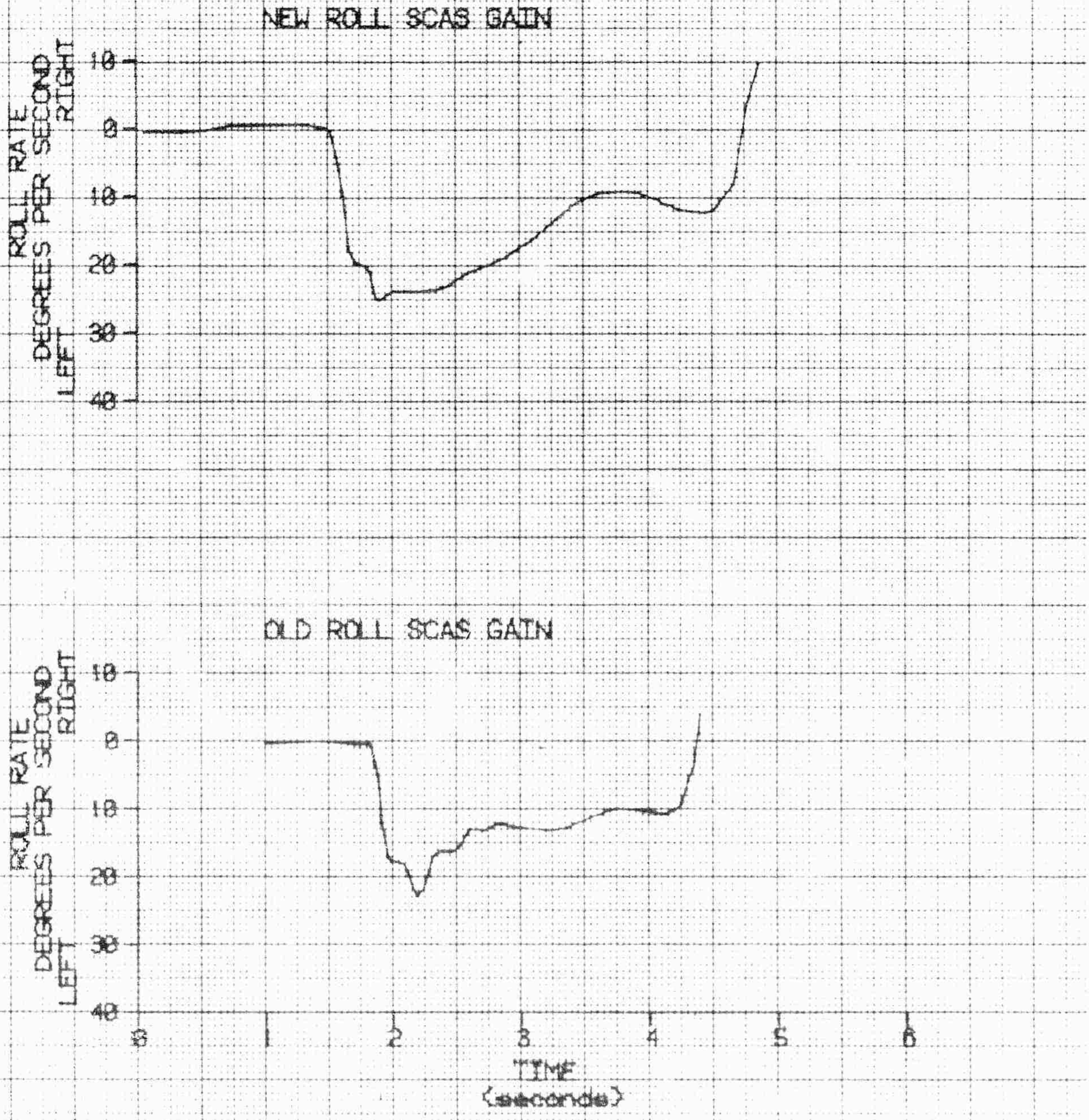


FIGURE 44
CONTROL SENSITIVITY AND DAMPING
OH-58D USA S/N 68-16285

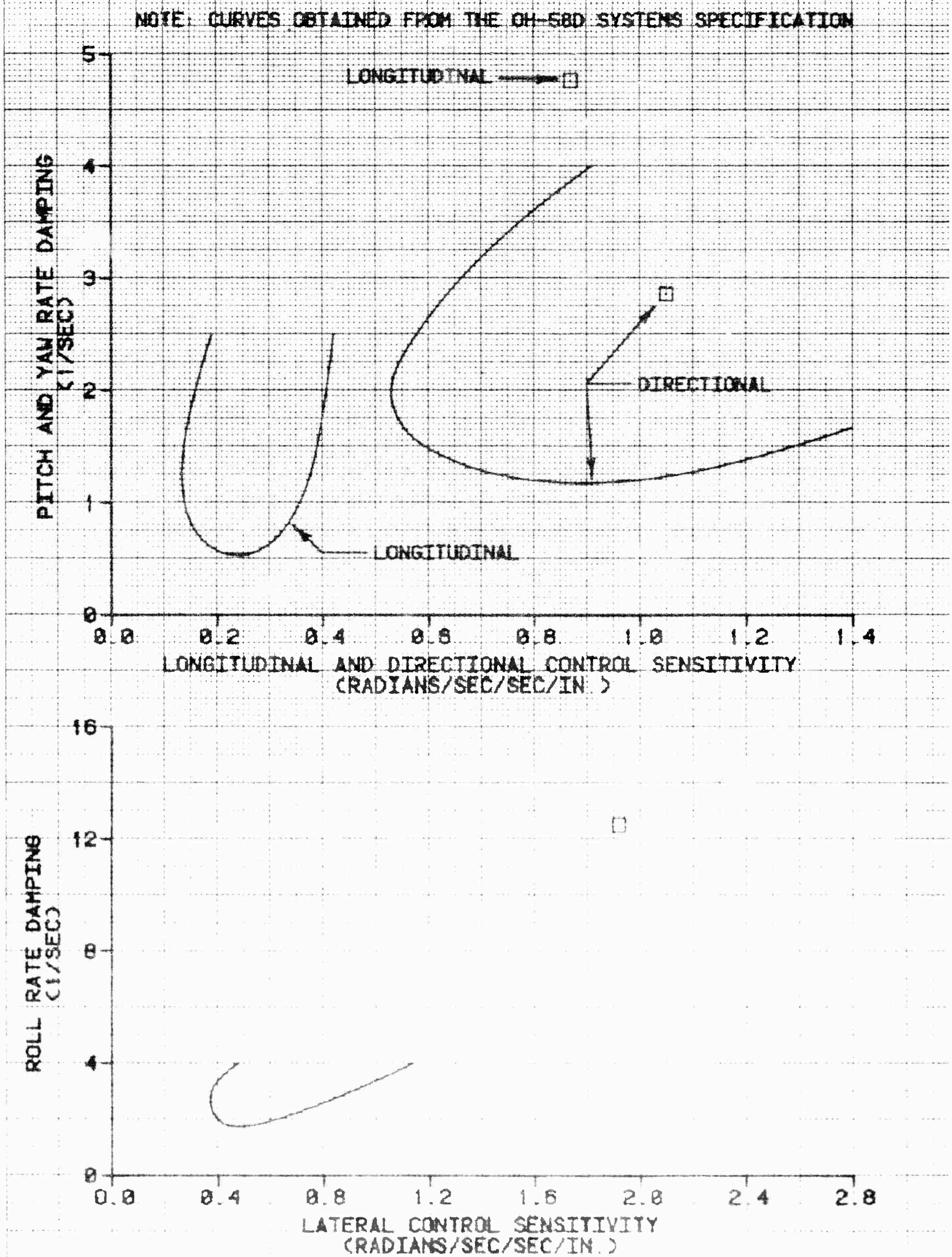


FIGURE 45
RIGHT 10 DEGREE SLOPE
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4500	111.30	1642	20.0	395	0	SLOPE LANDING

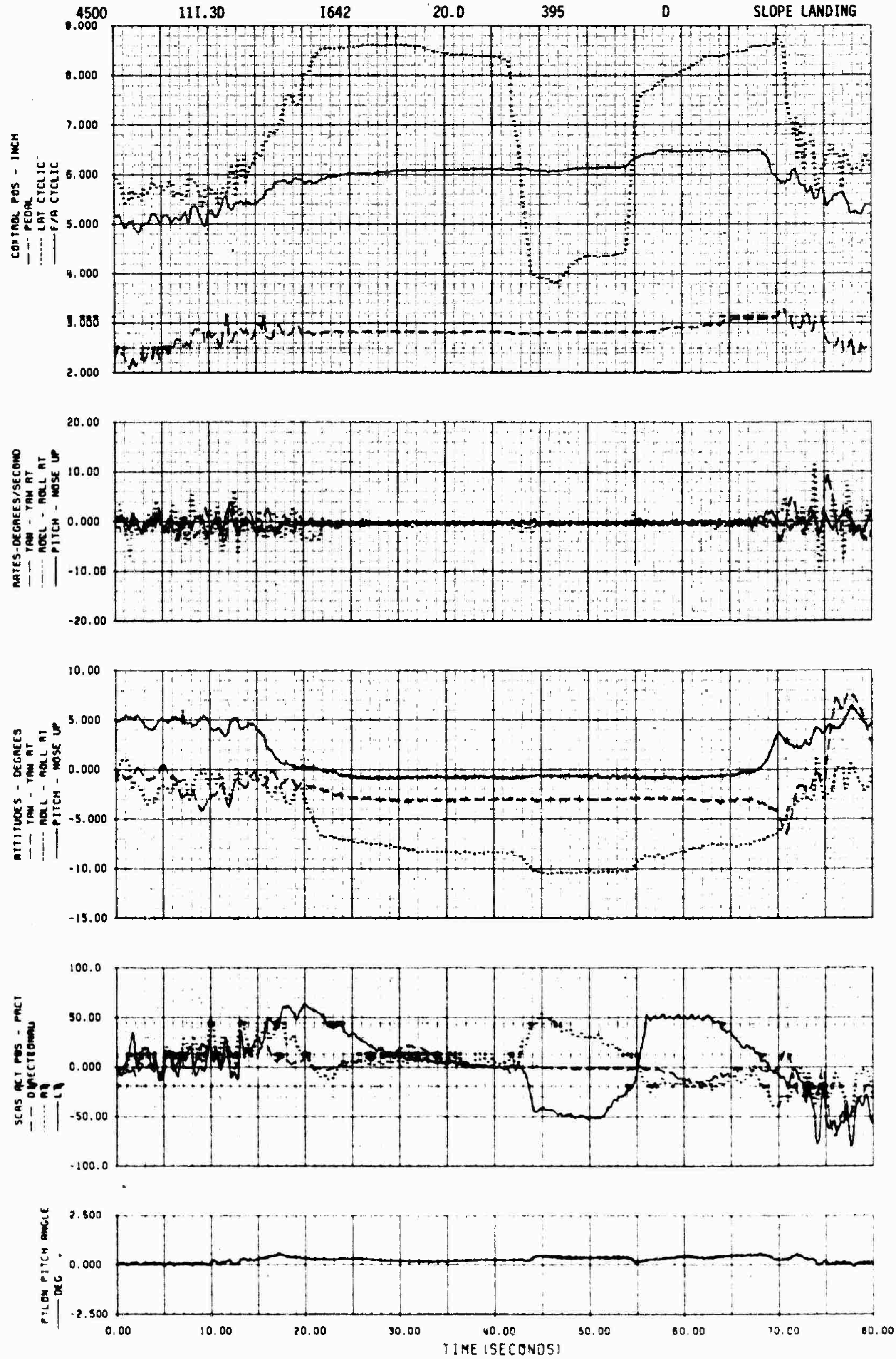


FIGURE 46A
LEFT 10 DEGREE SLOPE
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4608	110.60	1642	20.0	395	0	SLOPE LANDING

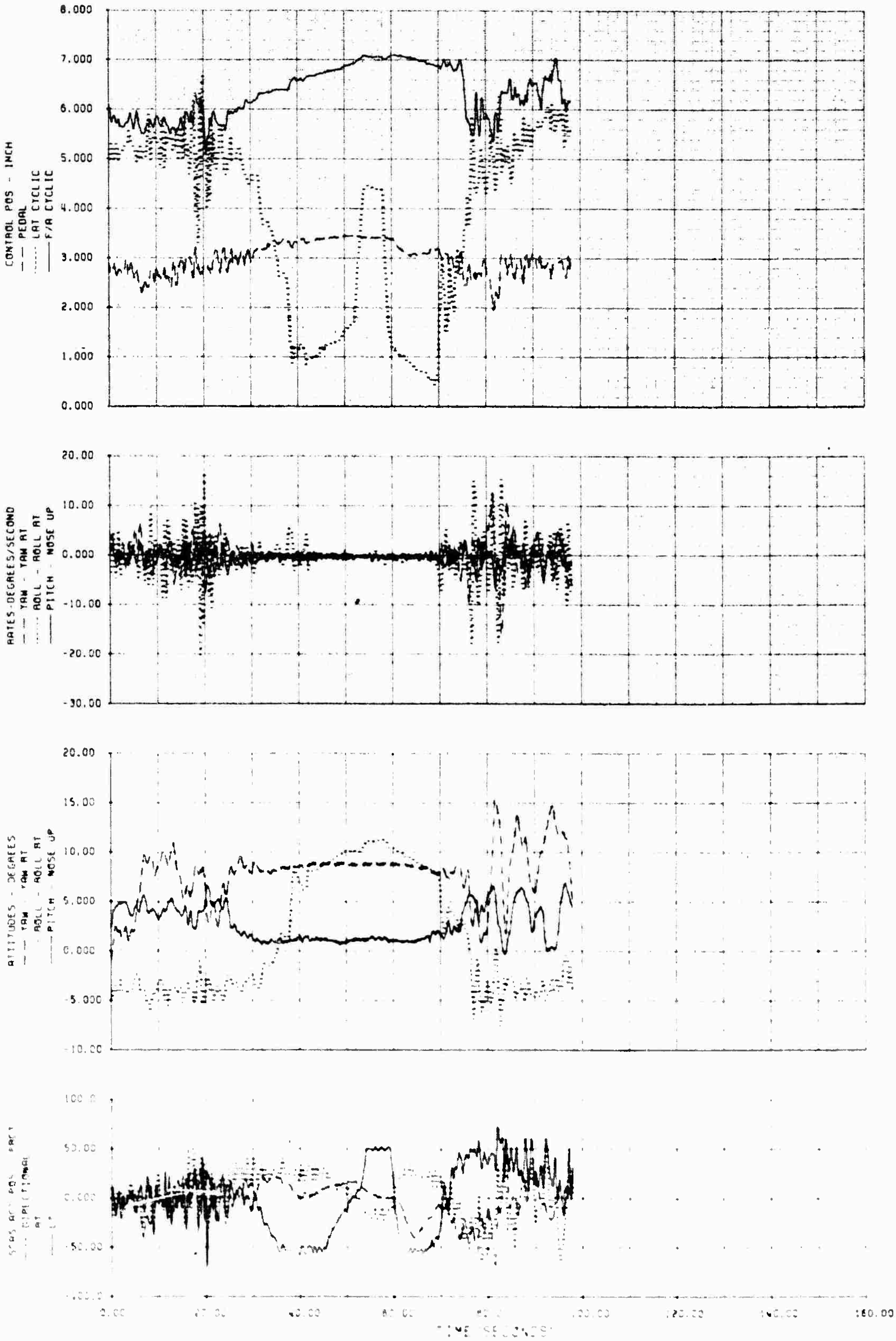


FIGURE 46B
LEFT 10 DEGREE SLOPE
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4608	110.60	1642	20.0	395	0	SLOPE LANDING

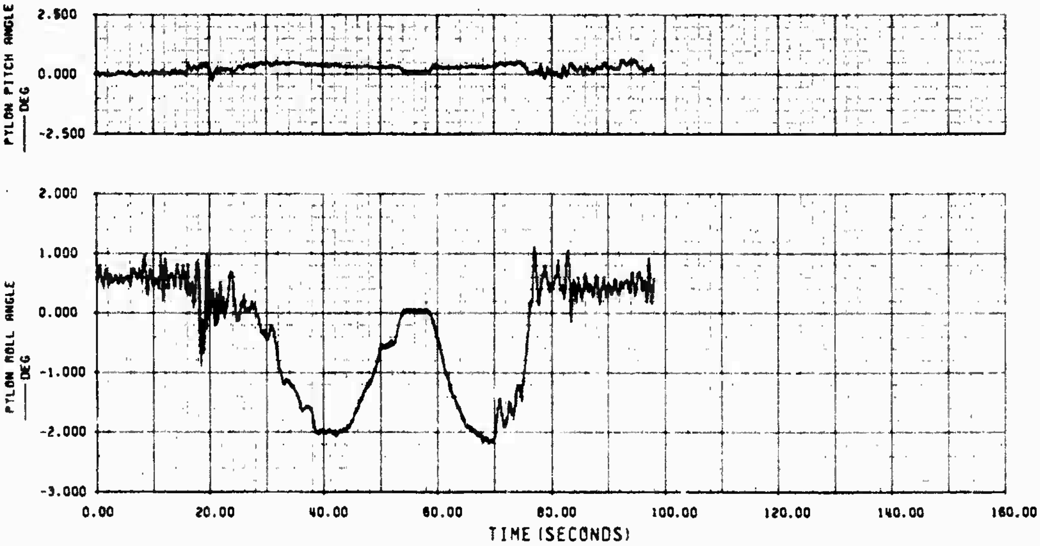


FIGURE 47

LOW SPEED FORWARD AND REARWARD FLIGHT

OH-58D USA S/N 69-16286

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	SKID HEIGHT
4400	110.9 (AFT)	880	15.0	395	10 FT

NOTES: 1. SCAS
2. WINDS LESS THAN 3 KNOTS
3. VERTICAL BARS DENOTE MAXIMUM CONTROL EXCURSIONS
4. EXTENDED HORIZONTAL STABILIZER INSTALLED

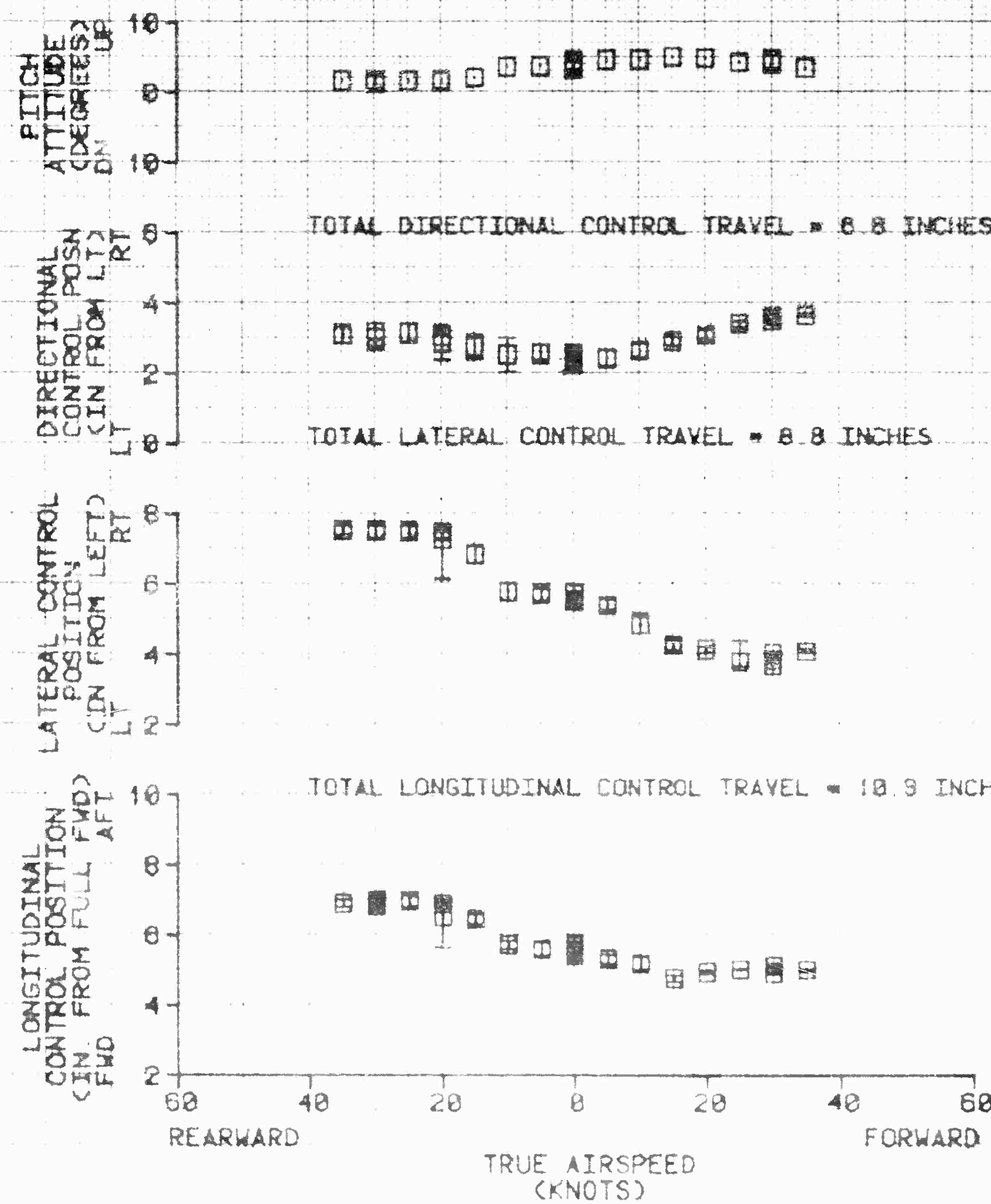


FIGURE 48
SIDEWARD FLIGHT
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	SKID HEIGHT
4490	110.5 (AFT)	560	14.0	395	10 FT

NOTES: 1. SCAS
2. WINDS LESS THAN 3 KNOTS
3. VERTICAL BARS DENOTE MAXIMUM CONTROL EXCURSIONS
4. EXTENDED HORIZONTAL STABILIZER INSTALLED

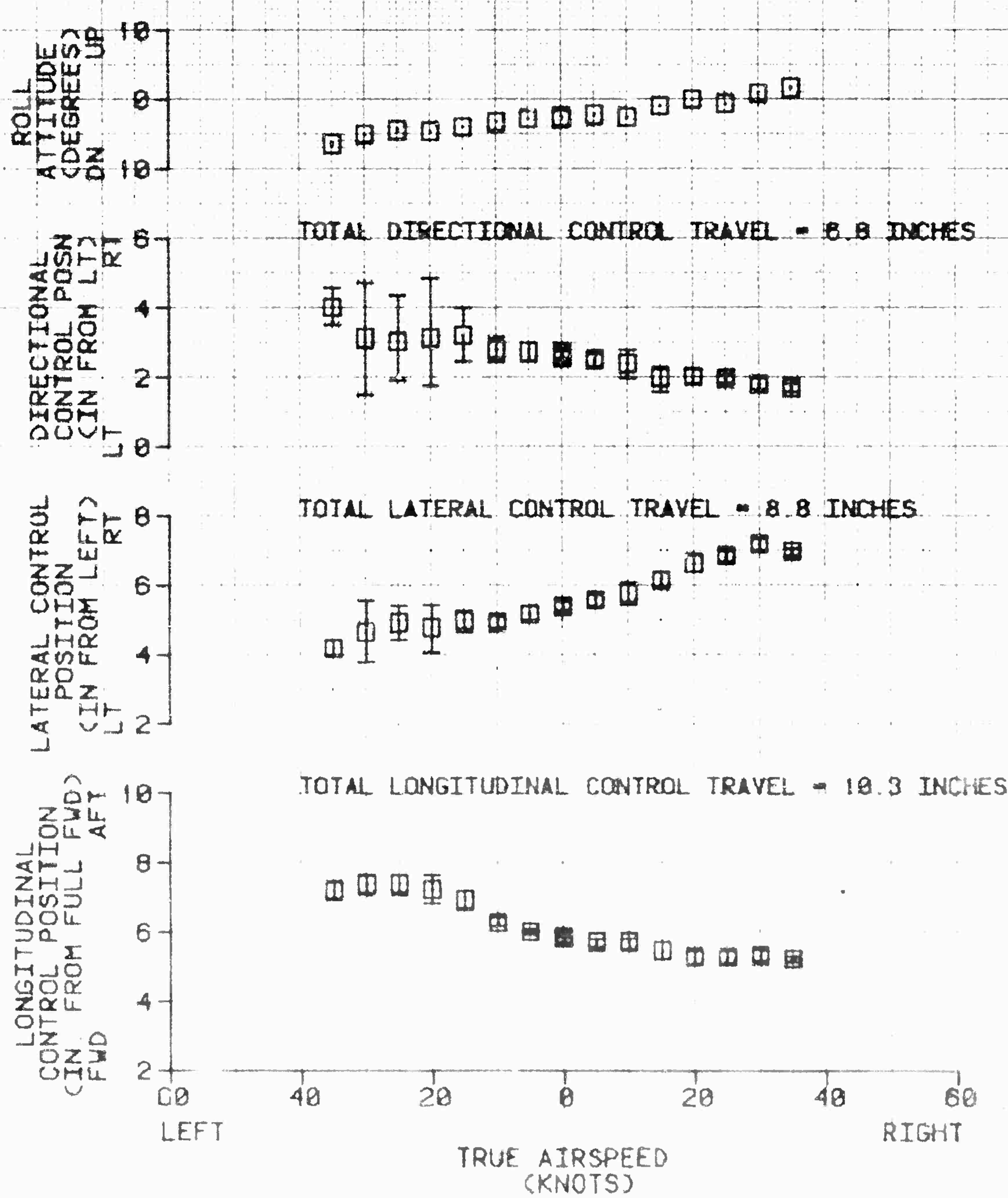


FIGURE 49A
DOWNWIND TURN TO HOVER
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4395	111.10	1830	18.5	395	50	LANDING

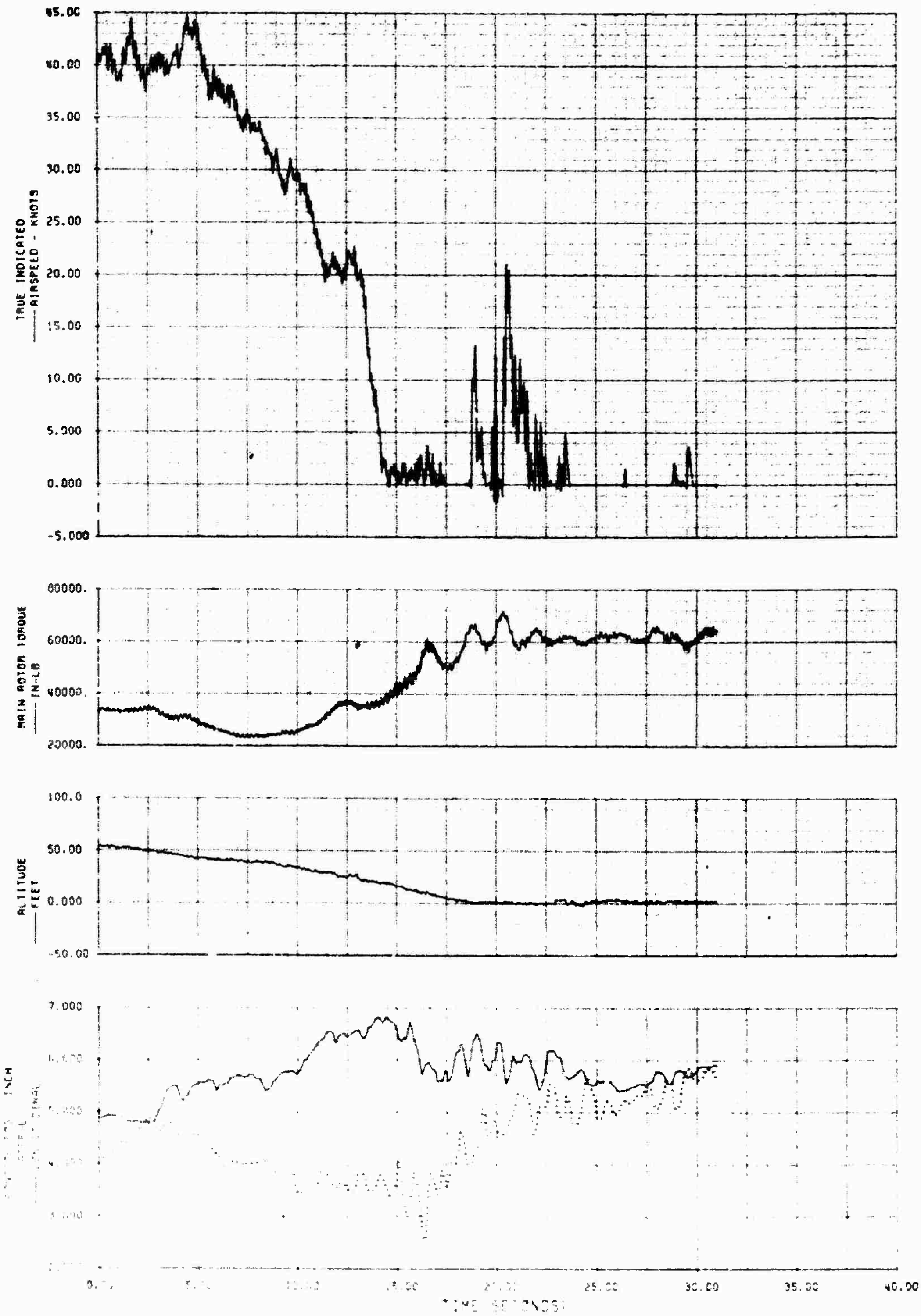


FIGURE 49B
DOWNWIND TURN TO HOVER
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4395	111.10	1830	18.5	395	50	LANDING

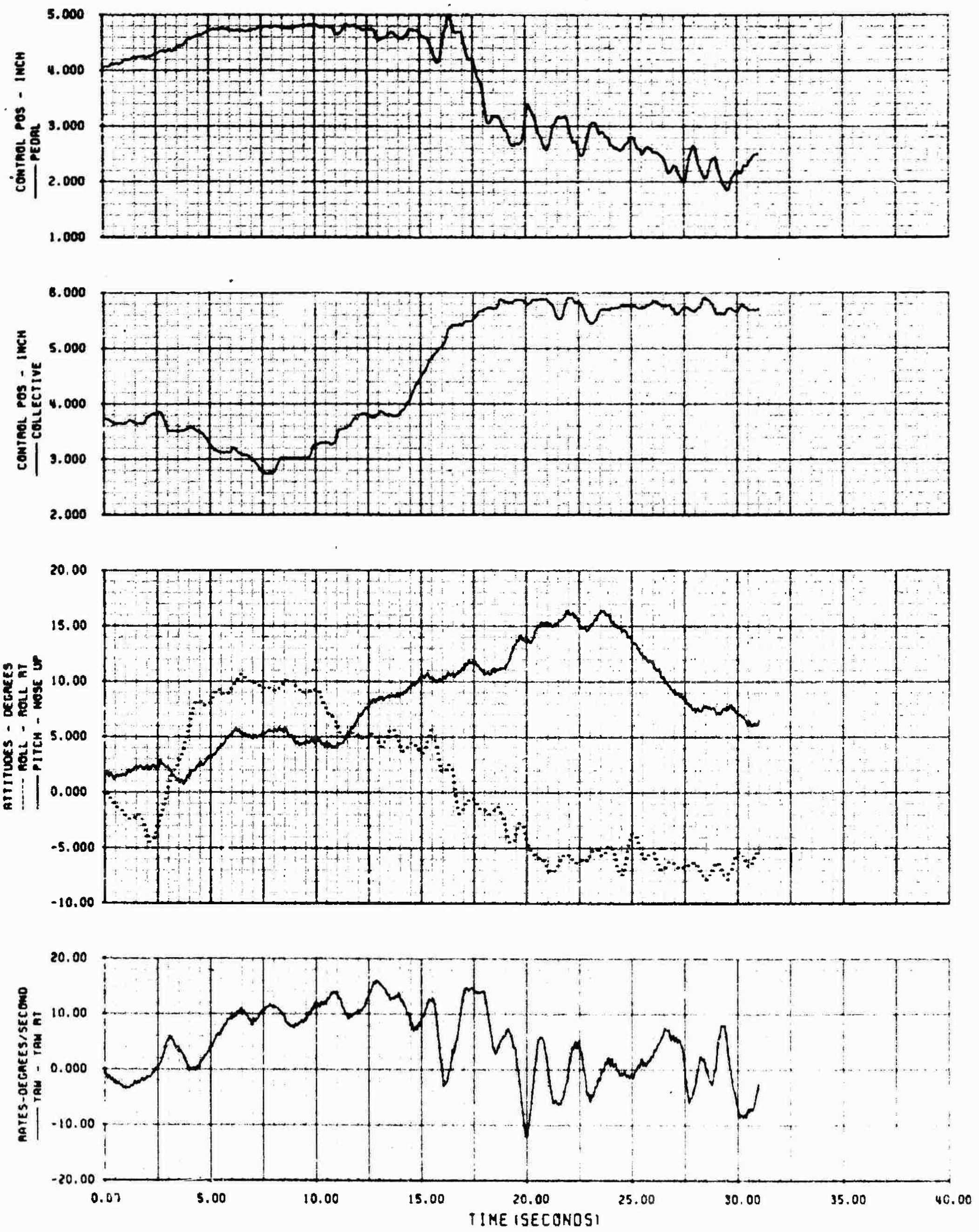


FIGURE 50A
RIGHT TURN TO LEFT CROSS WIND HOVER
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4395	111.10	1500	20.0	395	0	DESENT

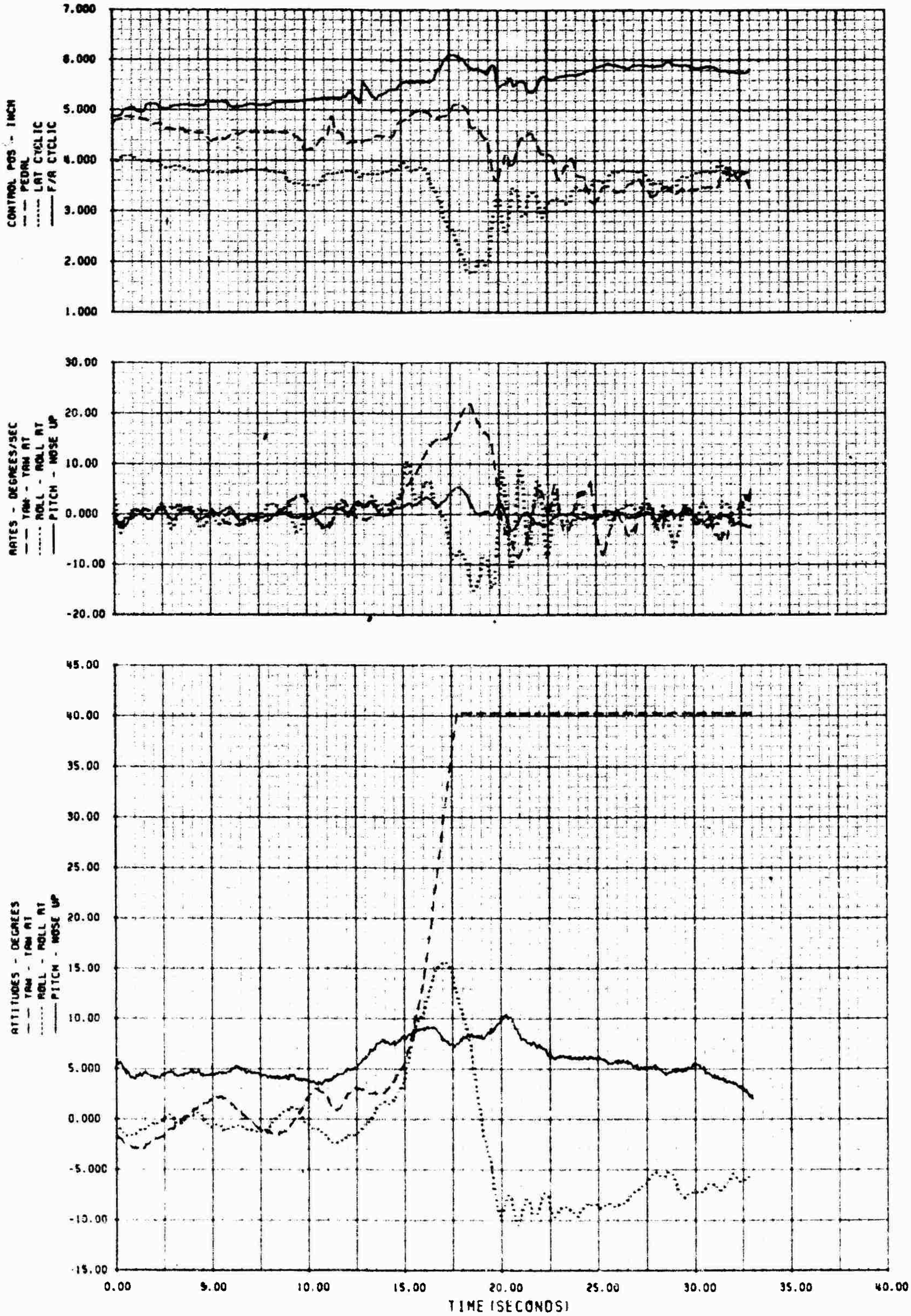


FIGURE 508
RIGHT TURN TO LEFT CROSS WIND HOVER
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4395	111.10	1500	20.0	395	0	DESENT

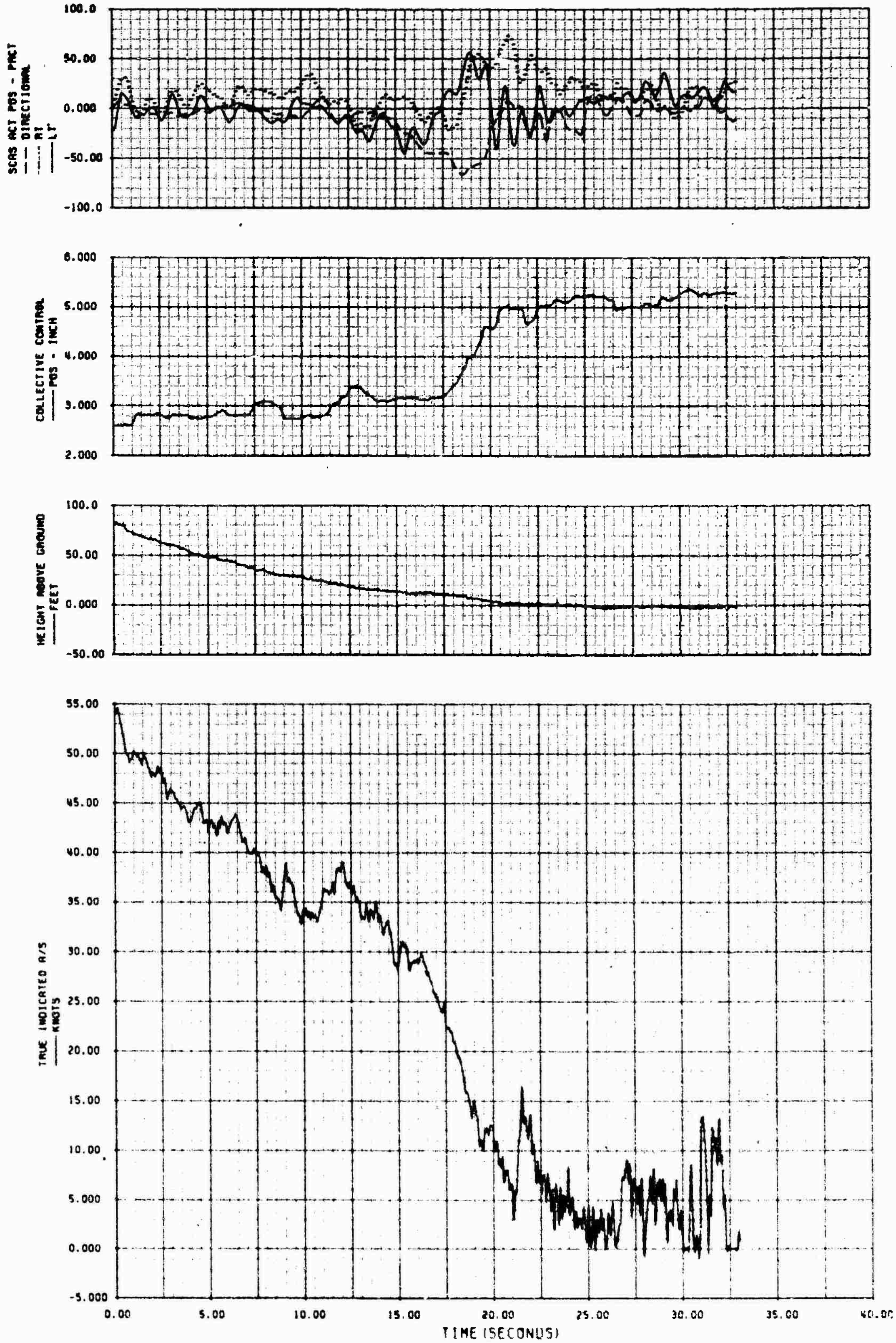


FIGURE 51A
SIMULATED ENGINE FAILURE
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4393	111.1	5776	14.5	395	105	LEVEL

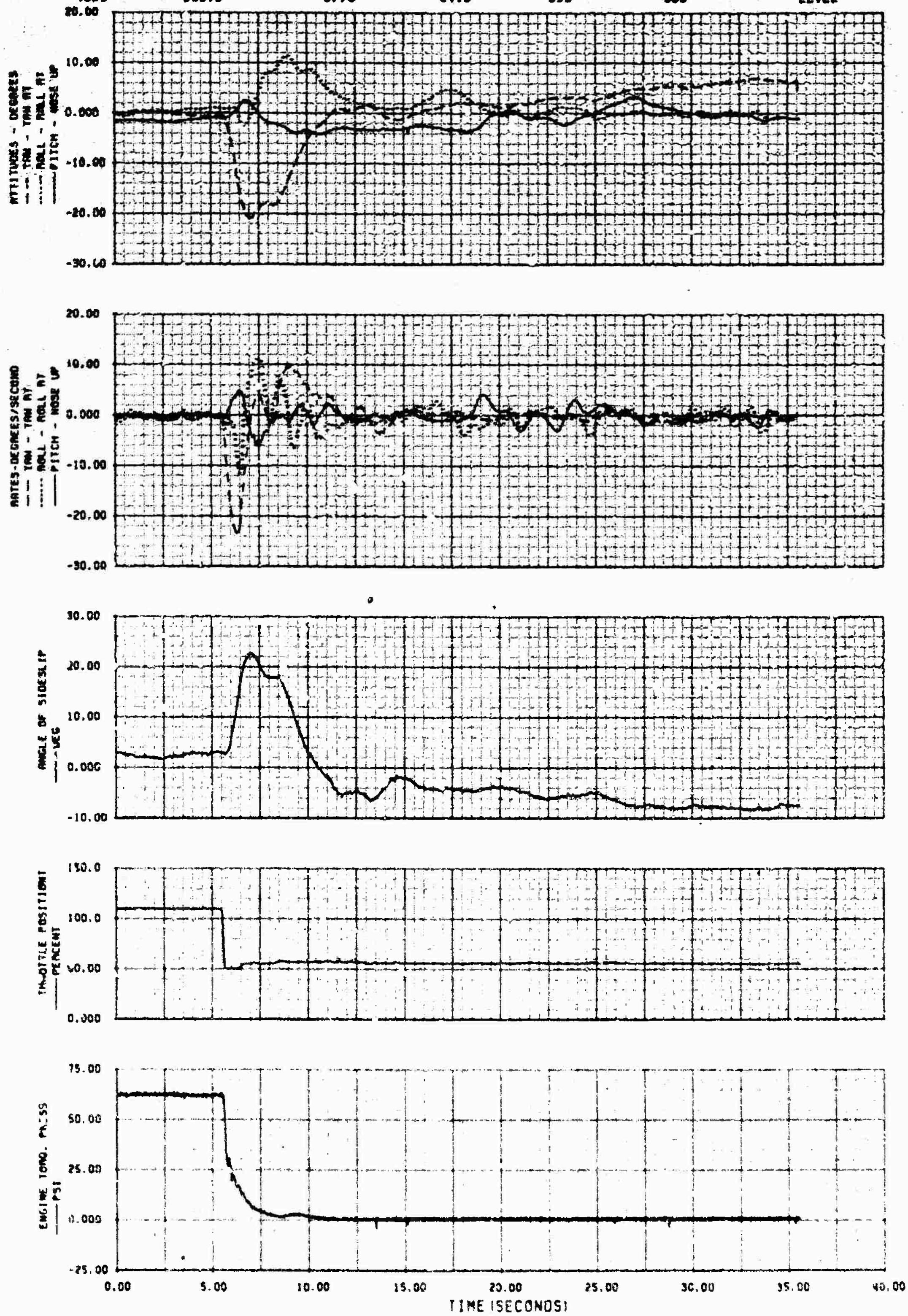


FIGURE 51B
SIMULATED ENGINE FAILURE
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4393	111.1	5776	14.5	396	106	LEVEL

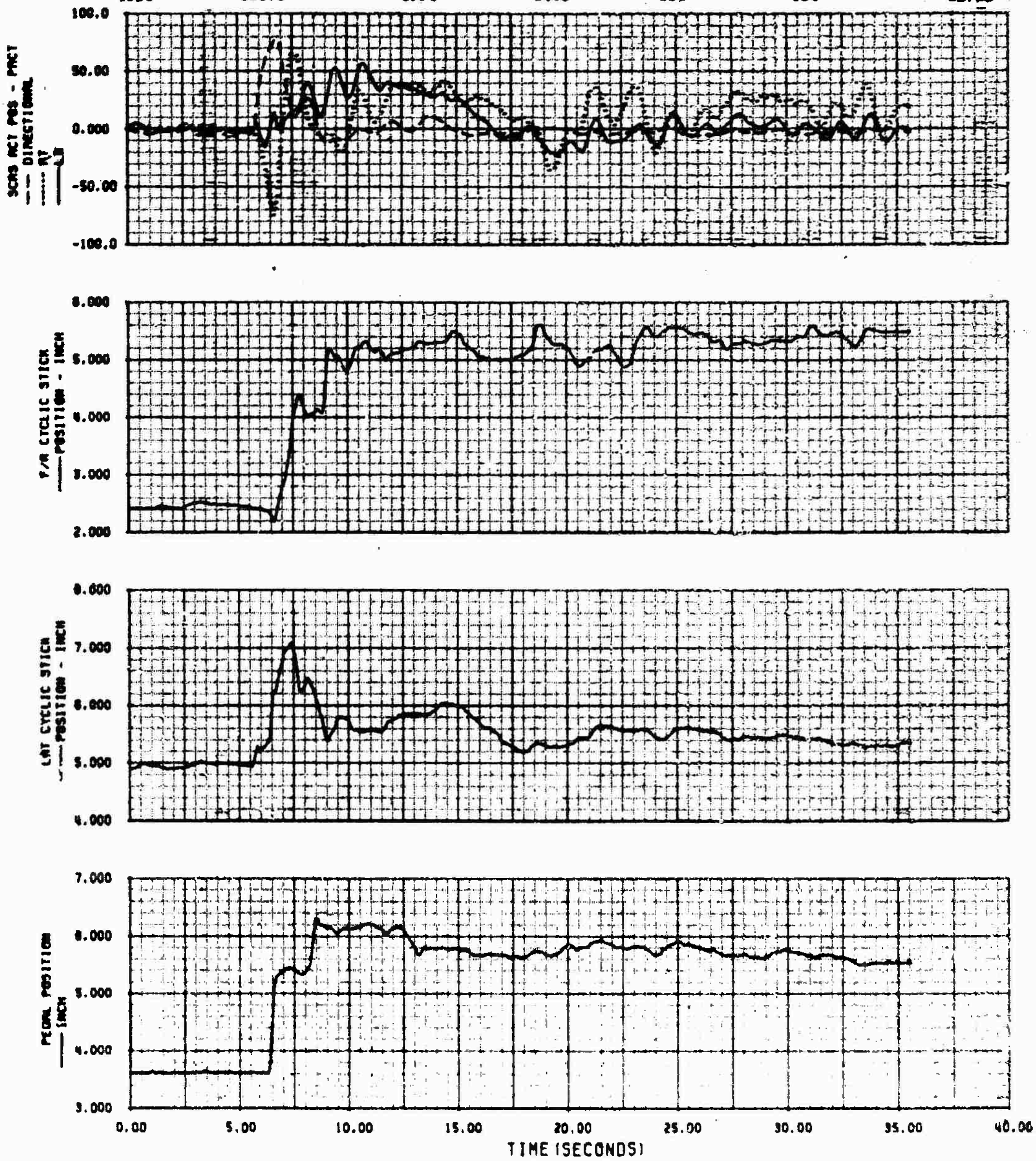


FIGURE 51C
SIMULATED ENGINE FAILURE
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4393	111.1	5776	14.5	395	109	LEVEL

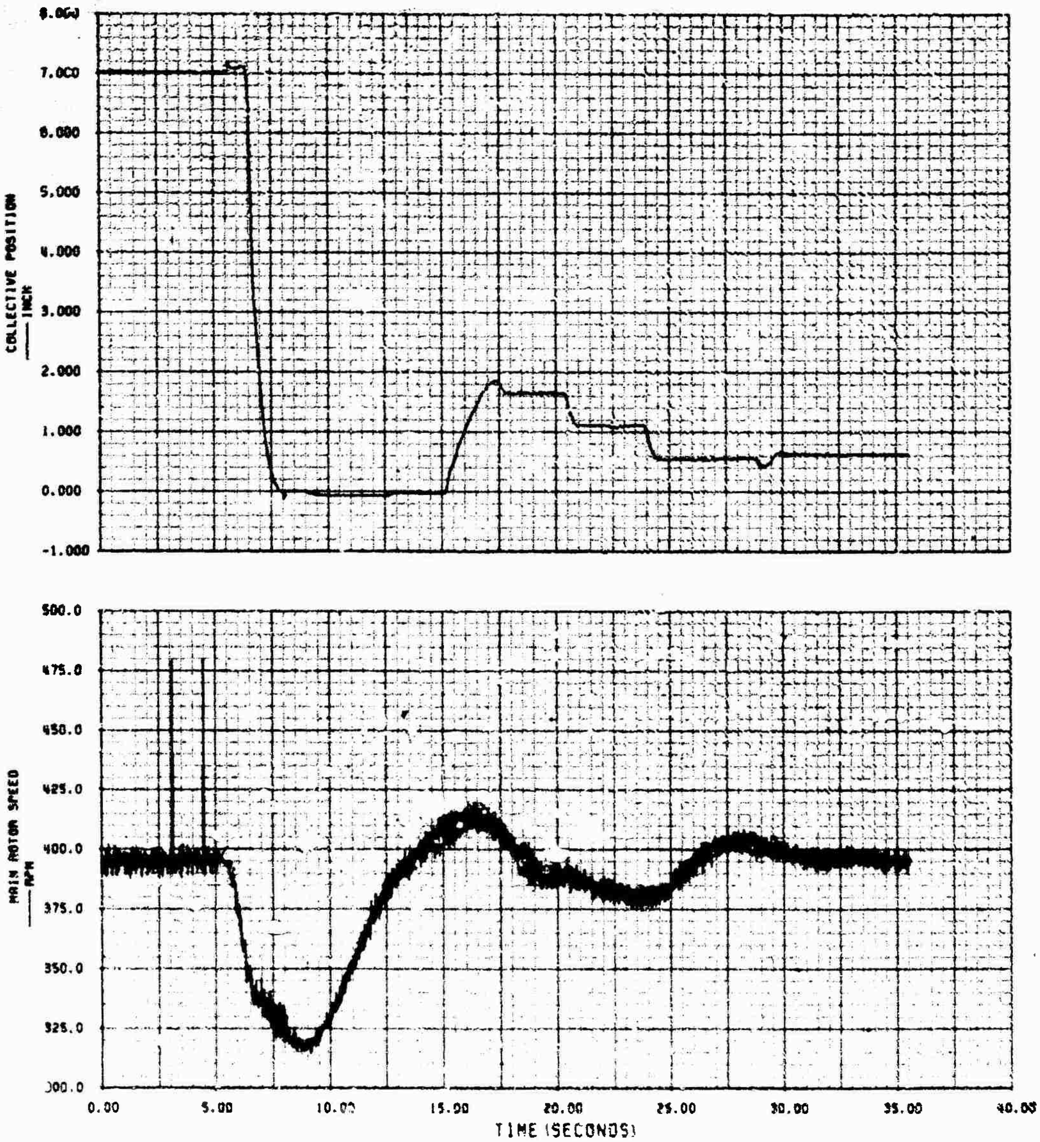


FIGURE 52
SCAS DISENGAGEMENT
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4394	111.8	5903	14.0	395	113	LEVEL FLIGHT

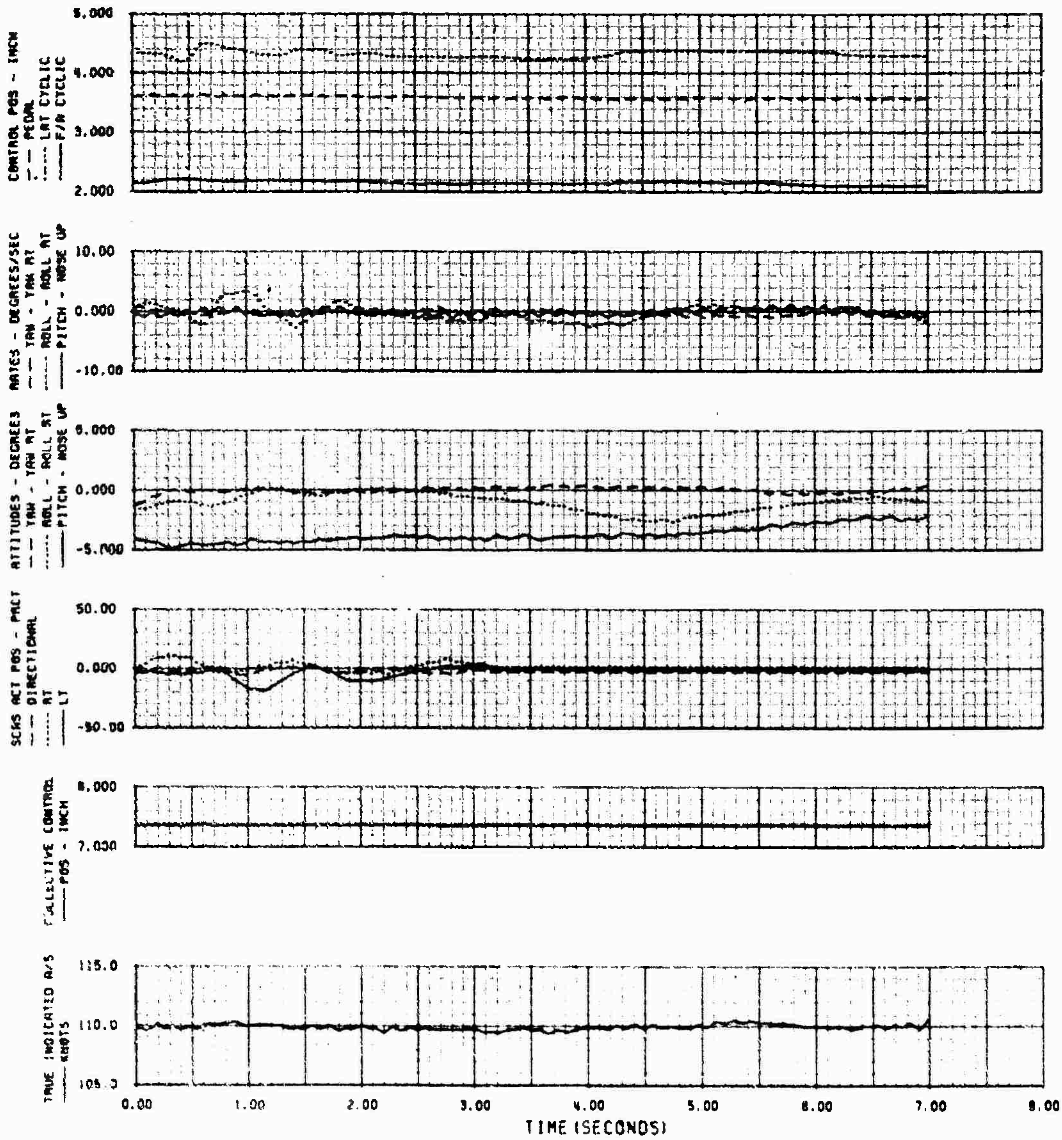


FIGURE 53
PITCH SCAS HARDOVER
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4394	110.8	5627	14.0	395	110	LEVEL FLIGHT

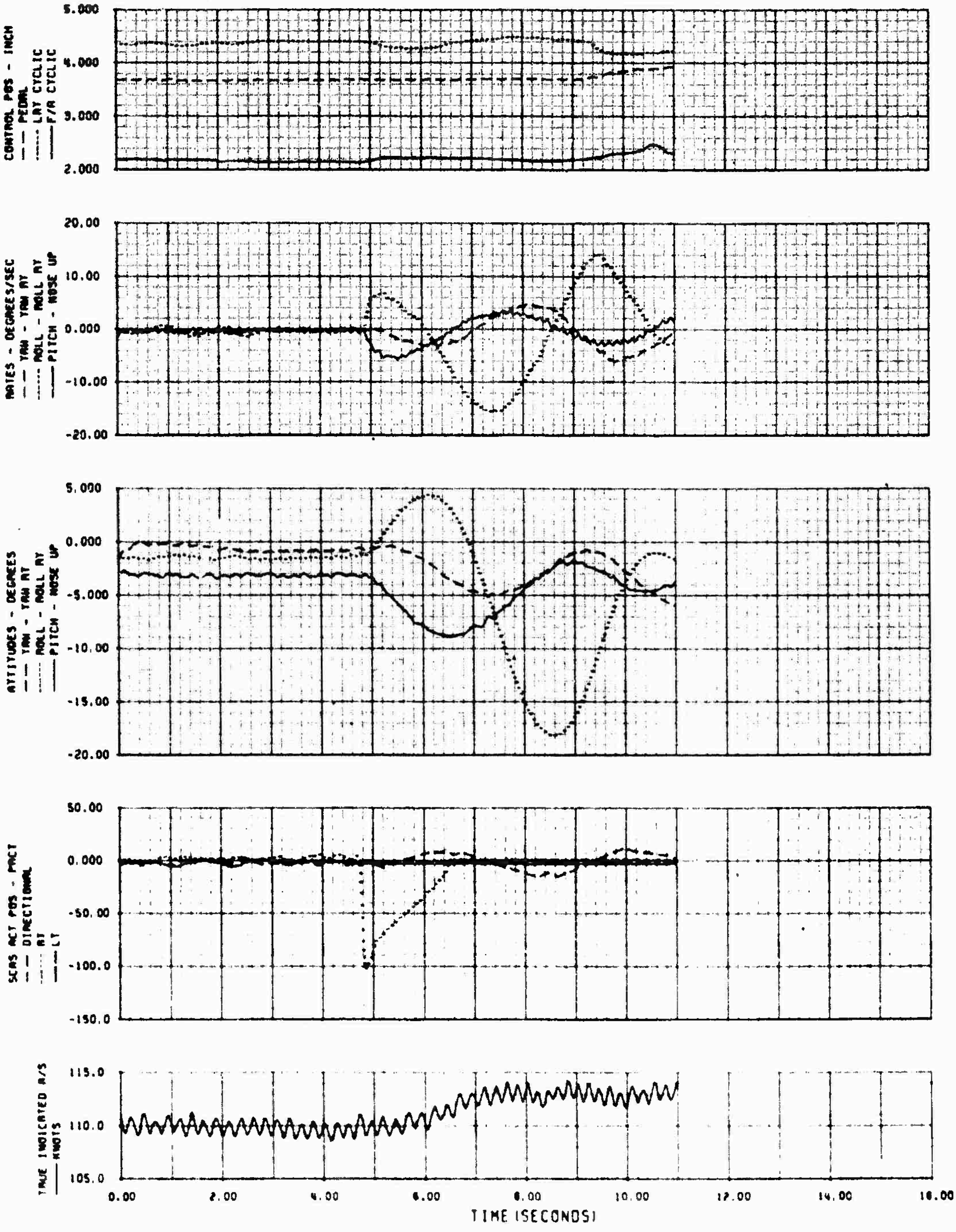


FIGURE 54A
DECELERATION FROM VNE TO 80 KIAS
(HYD OFF NORMAL STABILIZER)
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4394	111.8	6708	13.5	395	80	LEVEL FLIGHT

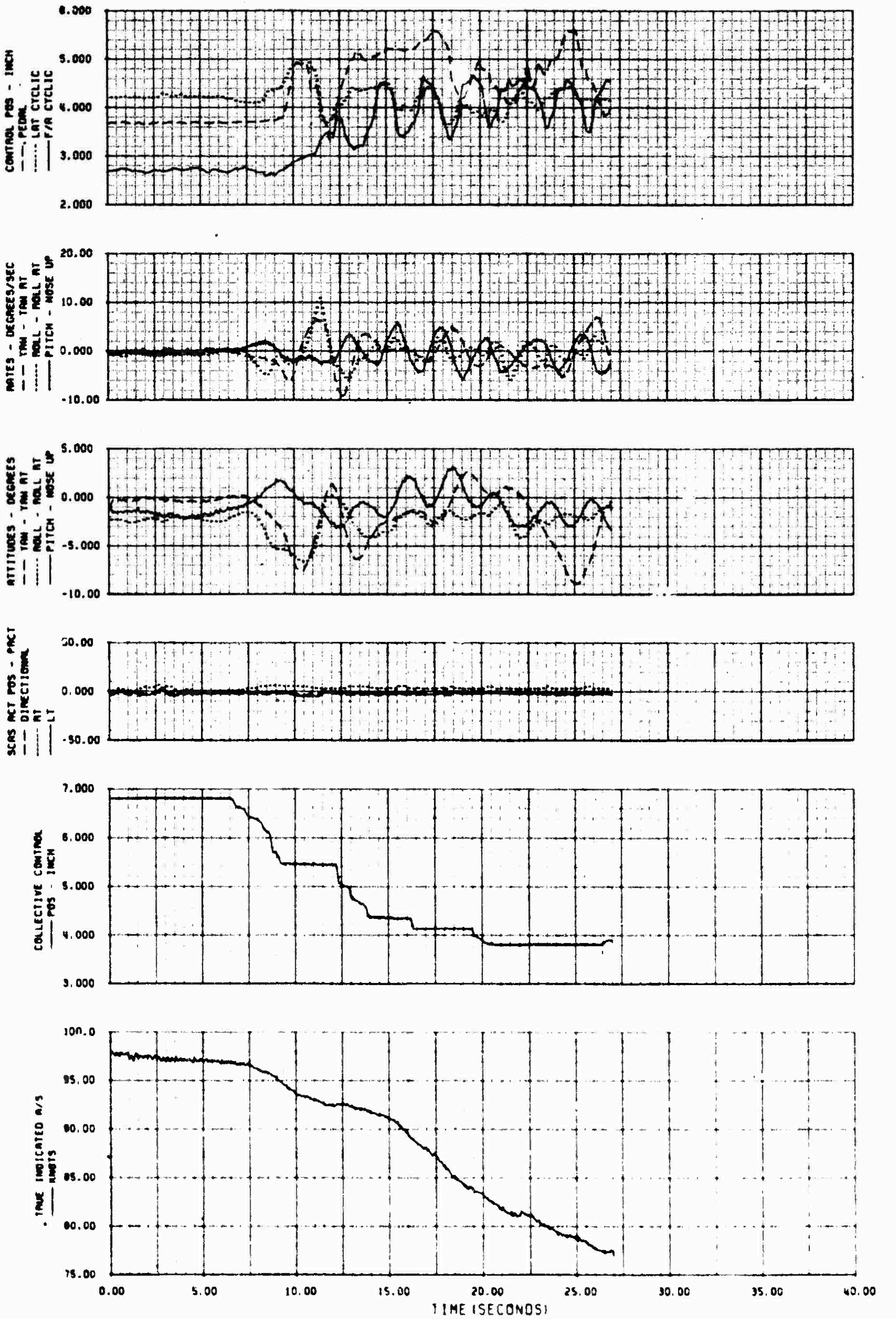


FIGURE 54B
DECELERATION FROM VNE TO 80 KIAS
(HYD OFF NORMAL STABILIZER)
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4394	111.8	6708	13.5	396	80	LEVEL FLIGHT

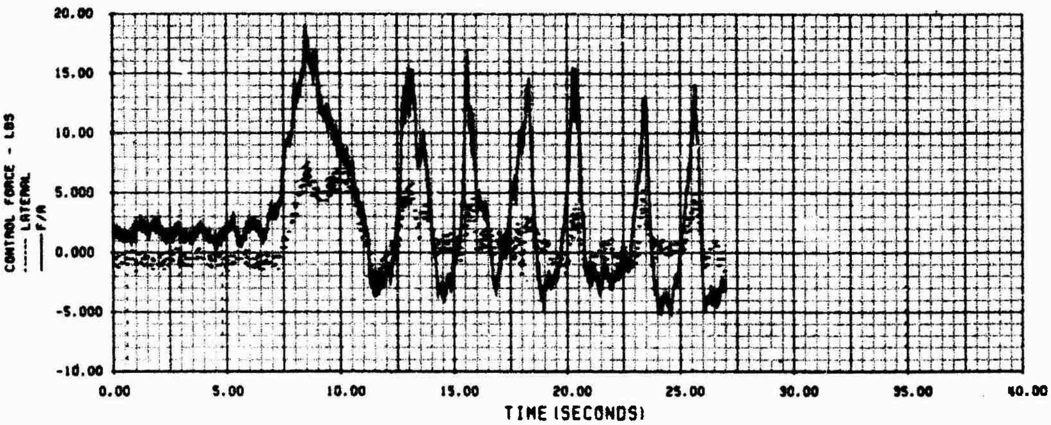


FIGURE 55A
DECELERATION FROM VNe TO 80 KIAS
(HYD OFF EXTENDED STABILIZER)
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4383	111.3	6729	13.5	395	105	LEVEL FLIGHT

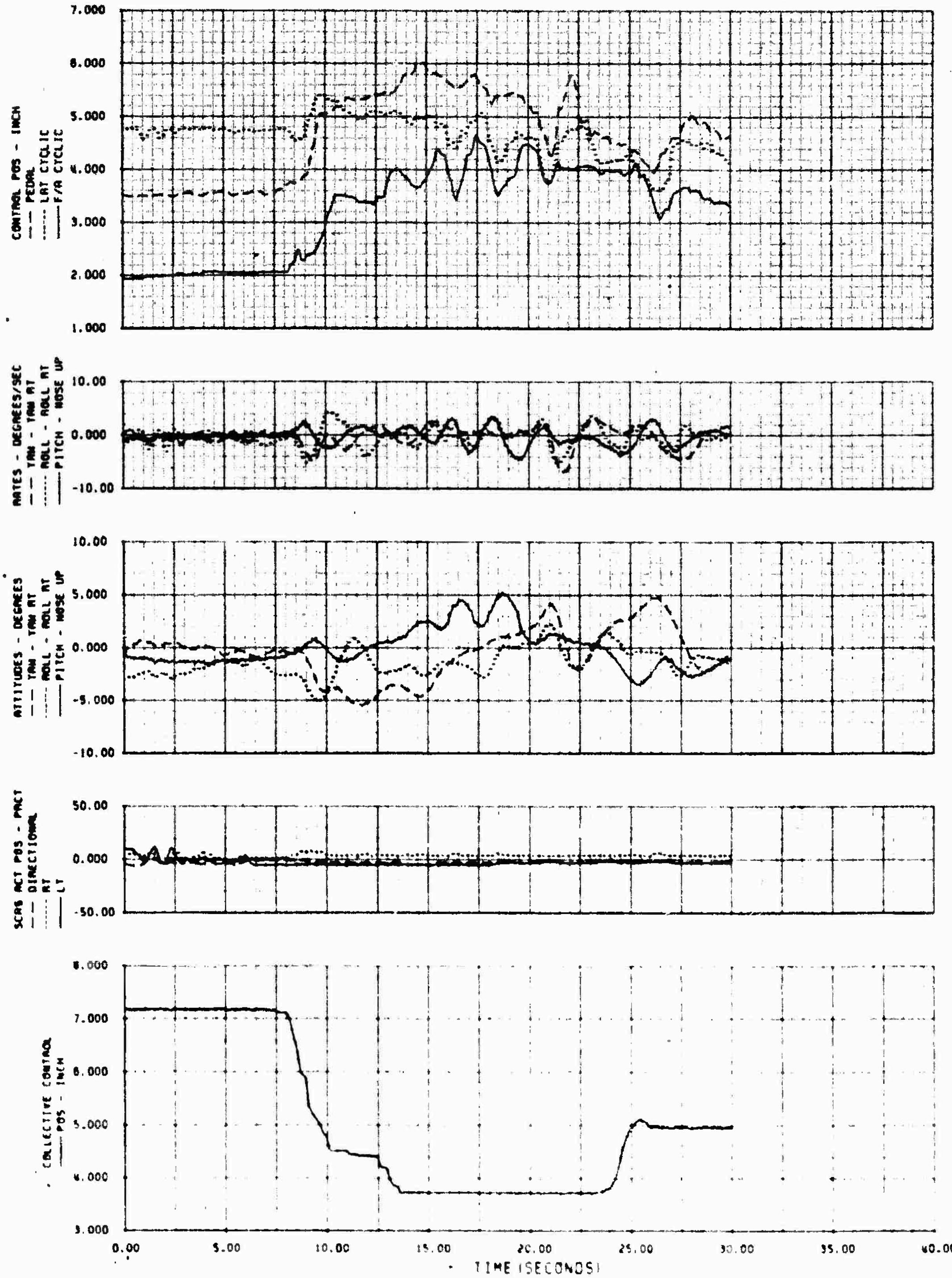


FIGURE 558
DECELERATION FROM VNe TO 80 KIAS
(HYD OFF EXTENDED STABILIZER)
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4383	111.3	6729	13.5	395	105	LEVEL FLIGHT

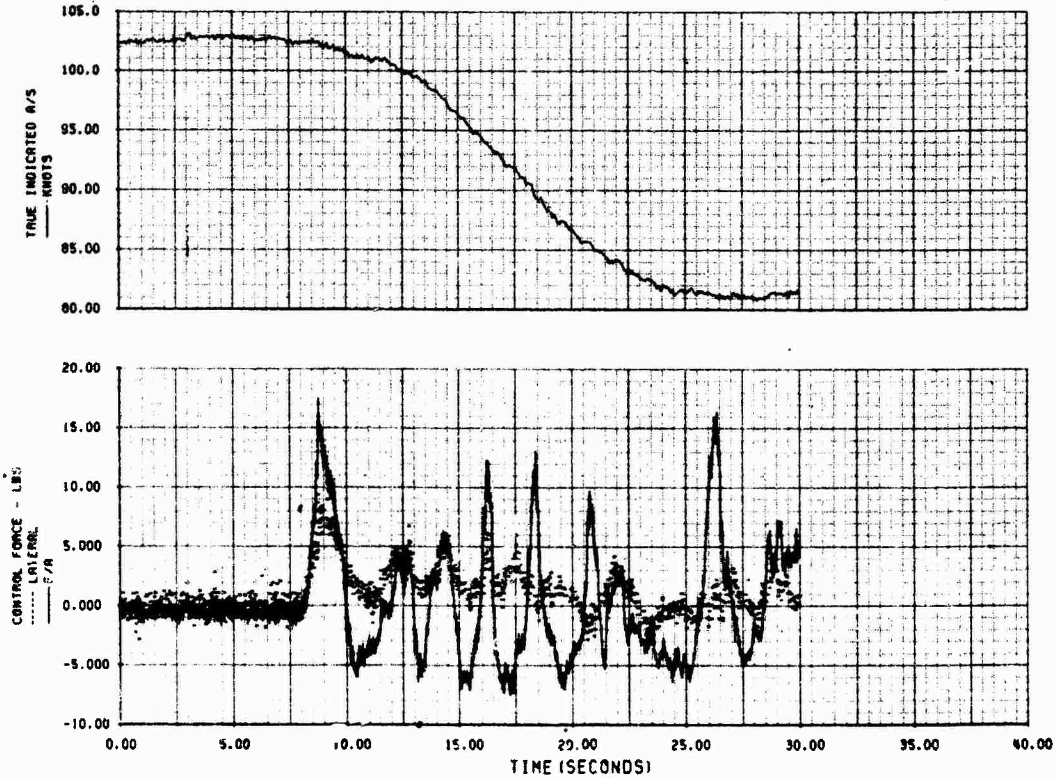


FIGURE 56A
AUTOROTATIONAL LANDING FROM A HOVER
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4084	111.80	487	24.0	395	0	HOVER

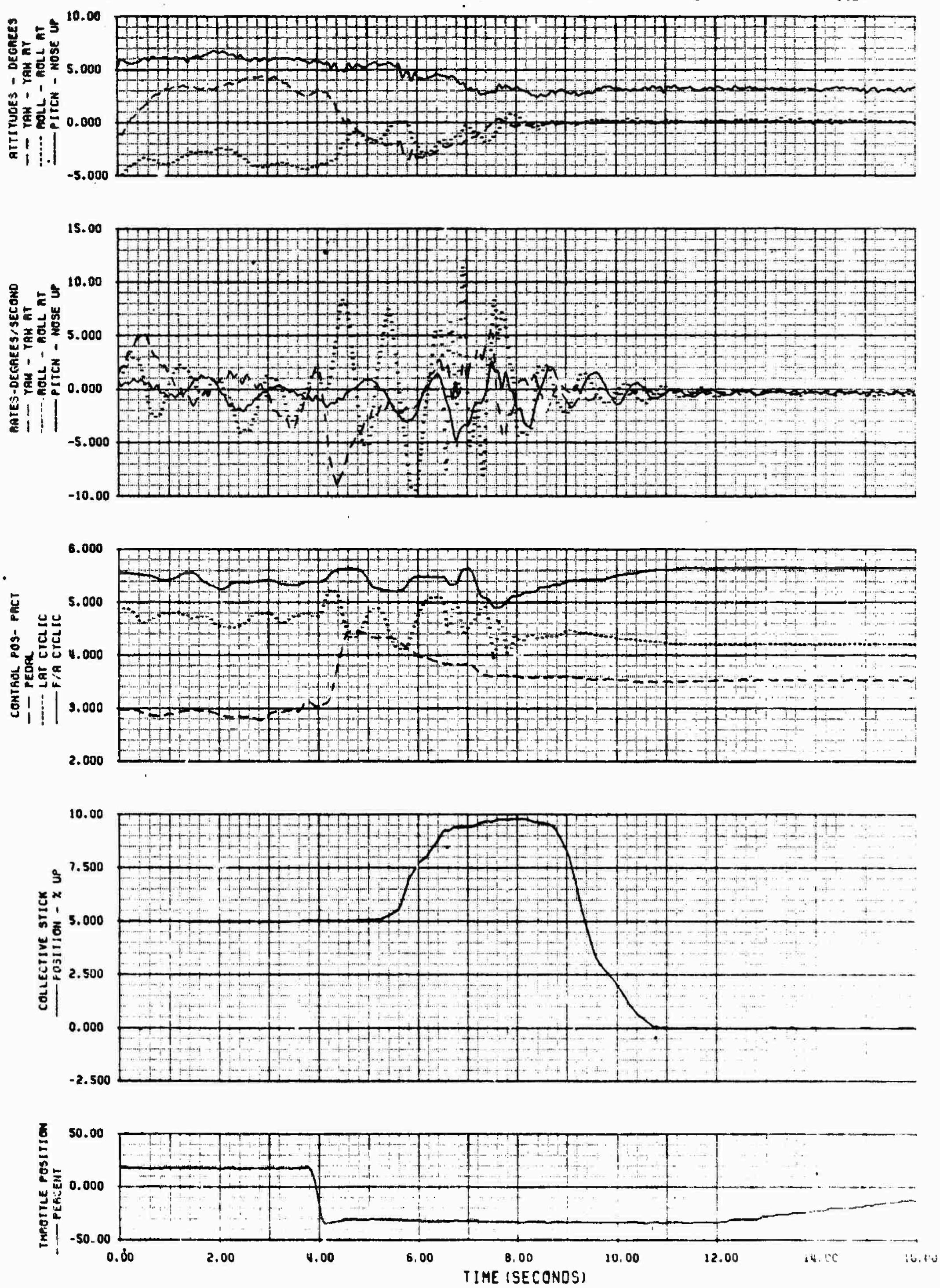


FIGURE 56B
AUTOROTATIONAL LANDING FROM A HOVER
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4084	111.80	487	24.0	395	0	HOVER

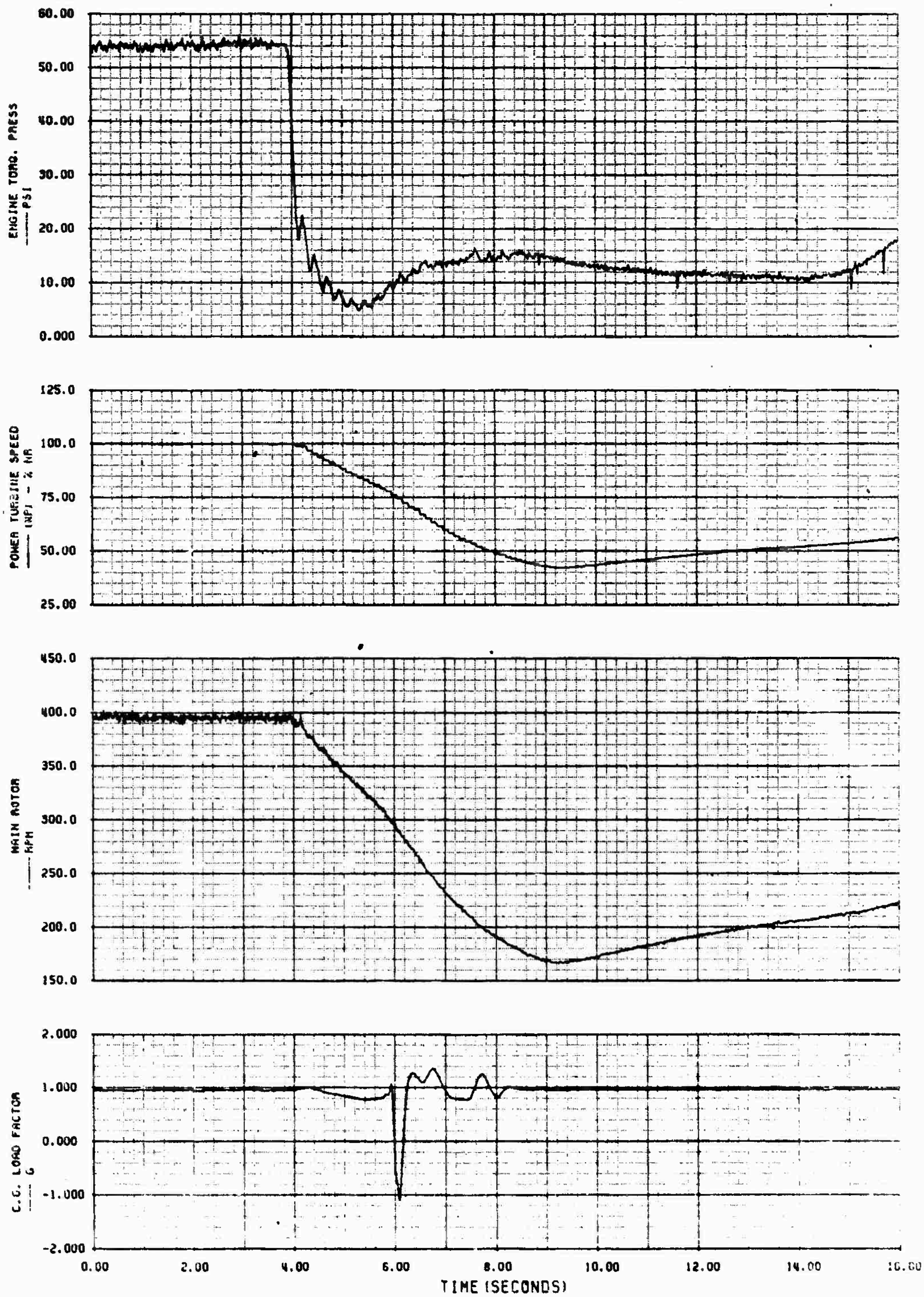


FIGURE 57A
AUTOROTATIONAL APPROACH AND LANDING
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4084	111.80	876	24.0	395	60	LEVEL FLIGHT

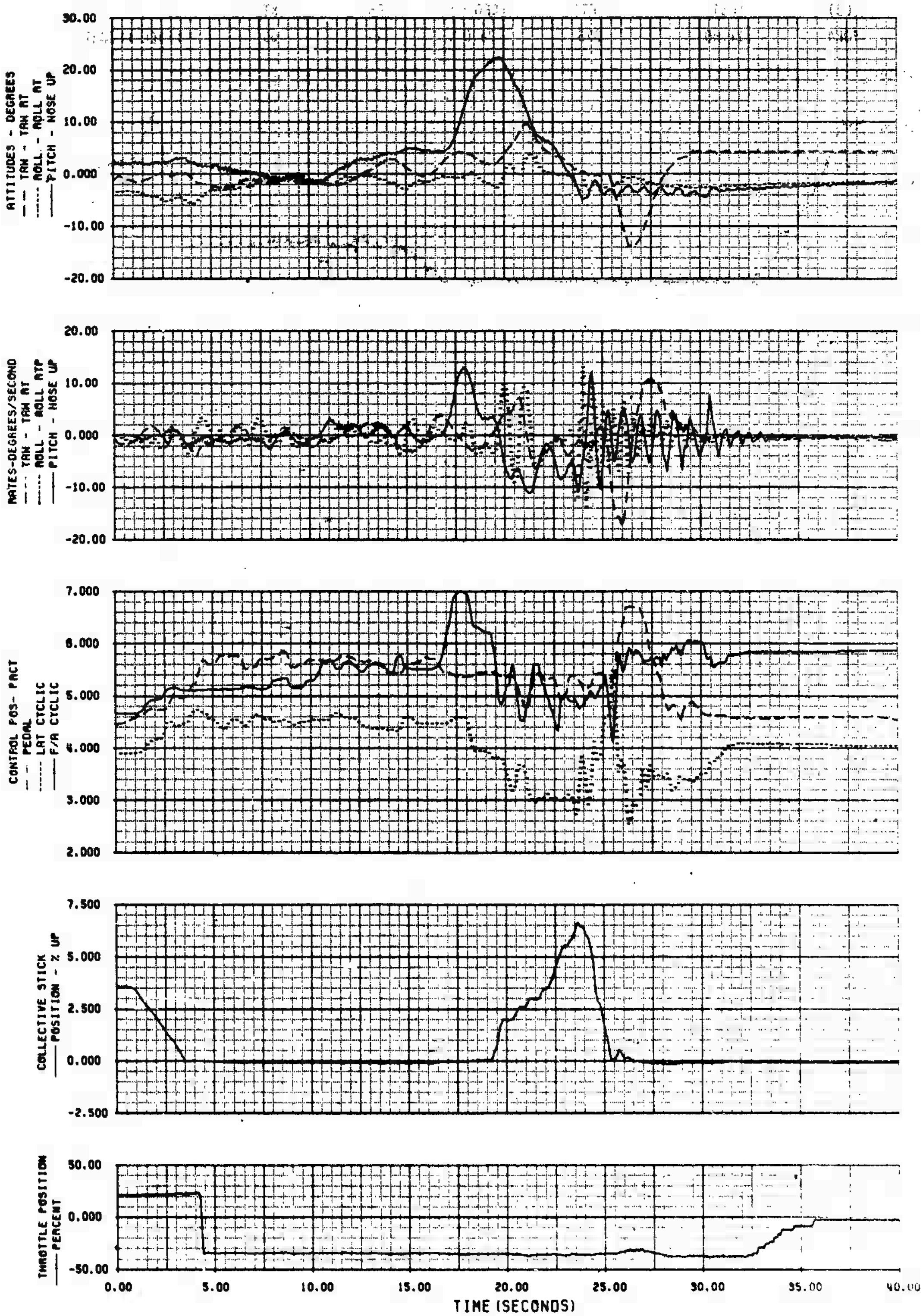


FIGURE 57B
AUTOROTATIONAL APPROACH AND LANDING
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4084	111.80	876	24.0	395	60	LEVEL FLIGHT

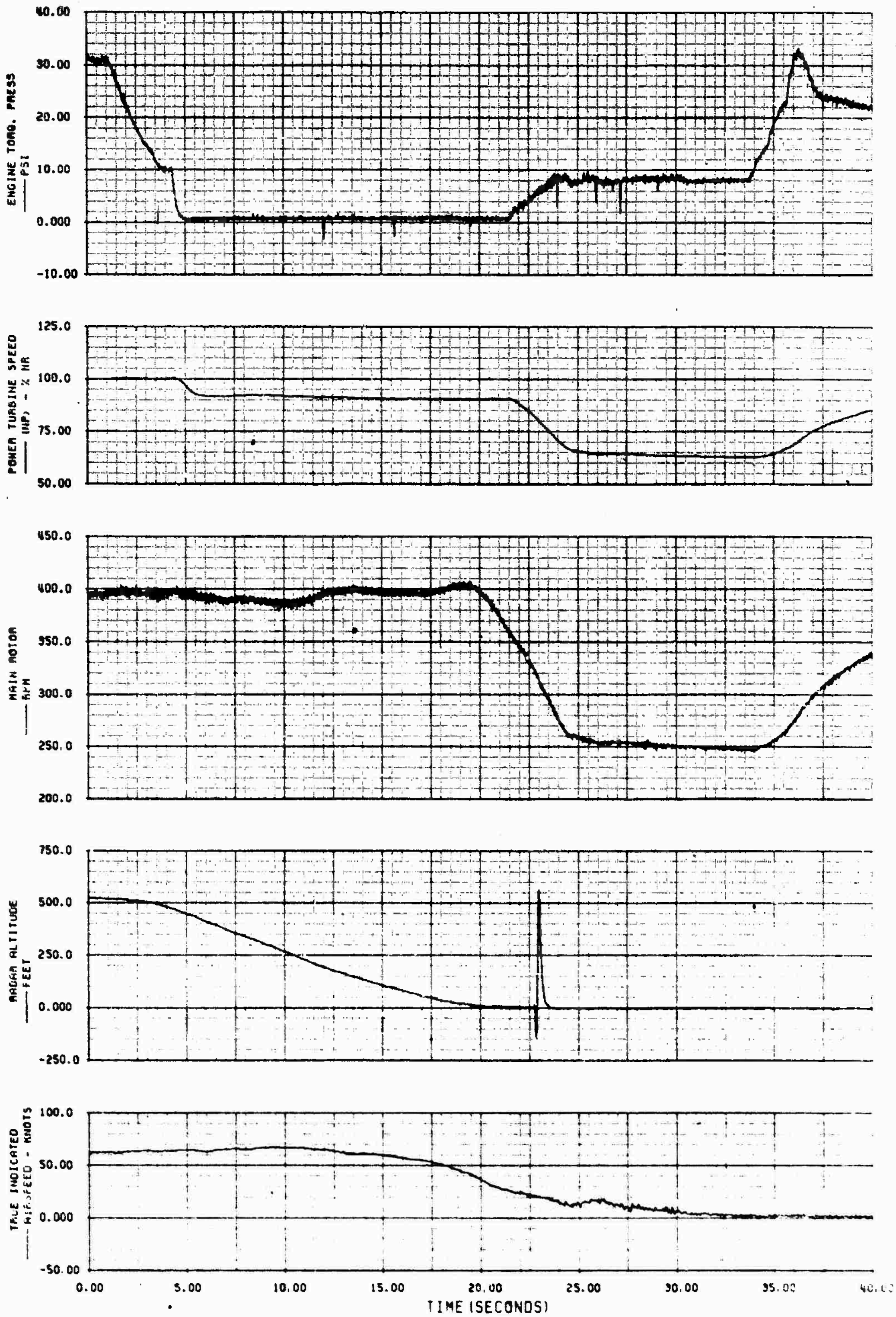


FIGURE 87C
AUTOROTATIONAL APPROACH AND LANDING
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4084	111.80	876	24.0	395	60	LEVEL FLIGHT

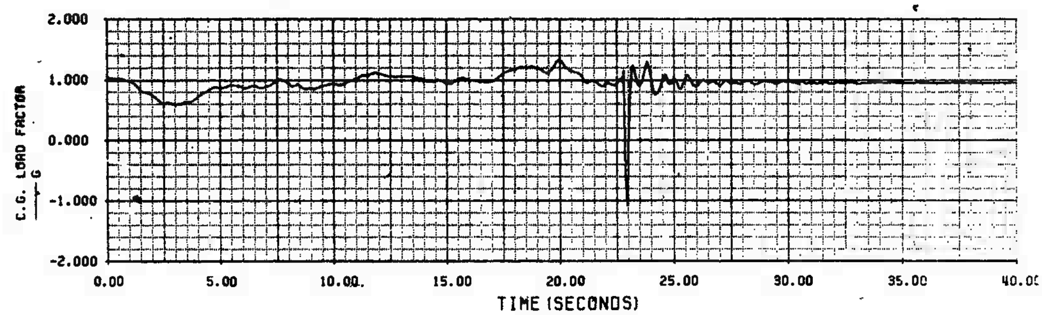


FIGURE 58A
AUTOROTATIONAL APPROACH AND LANDING
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4328	110.9	1609	22.0	396	55	AUTOROTATION

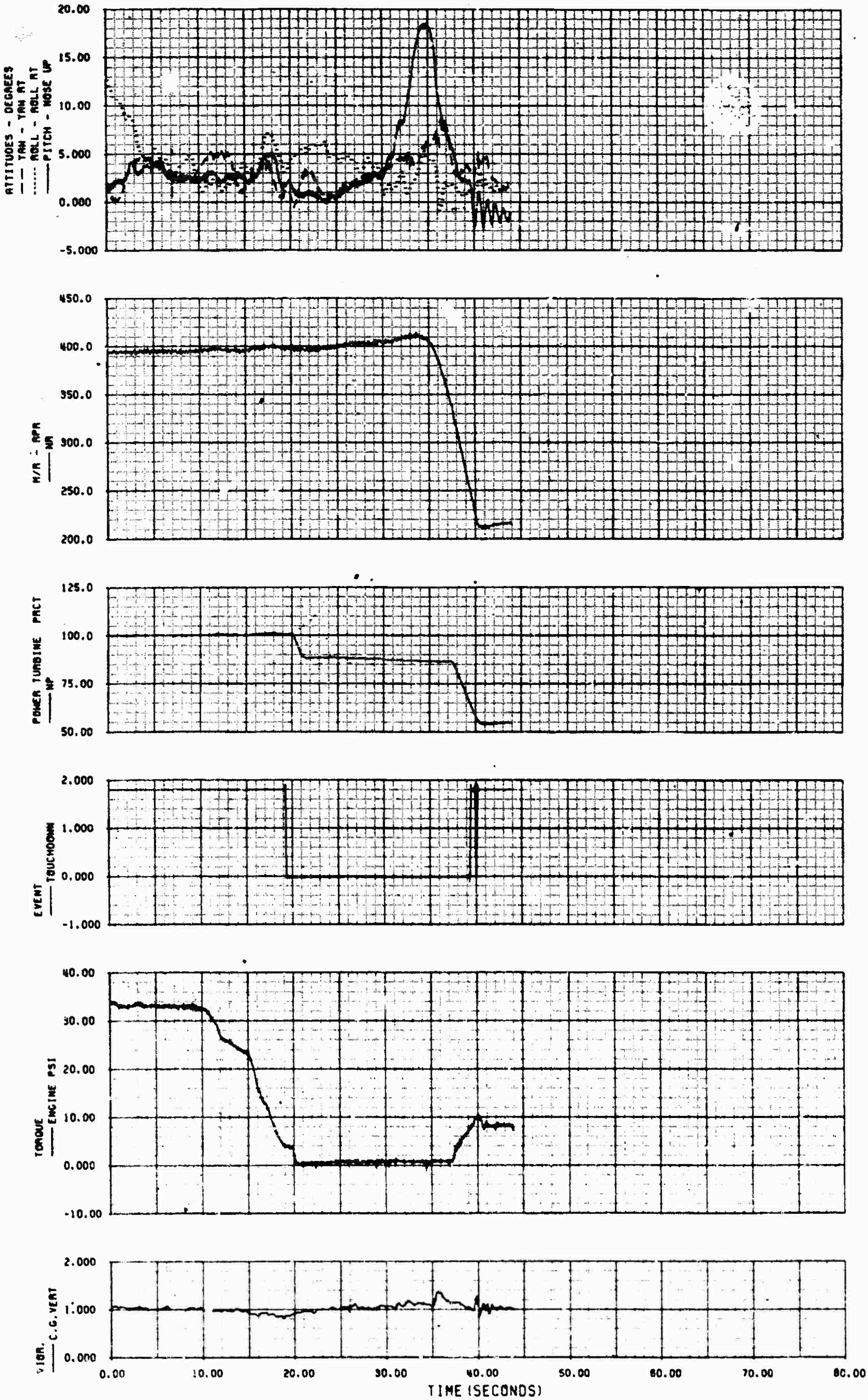


FIGURE 58B
AUTOROTATIONAL APPROACH AND LANDING
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4328	110.9	1609	22.0	396	55	AUTOROTATION

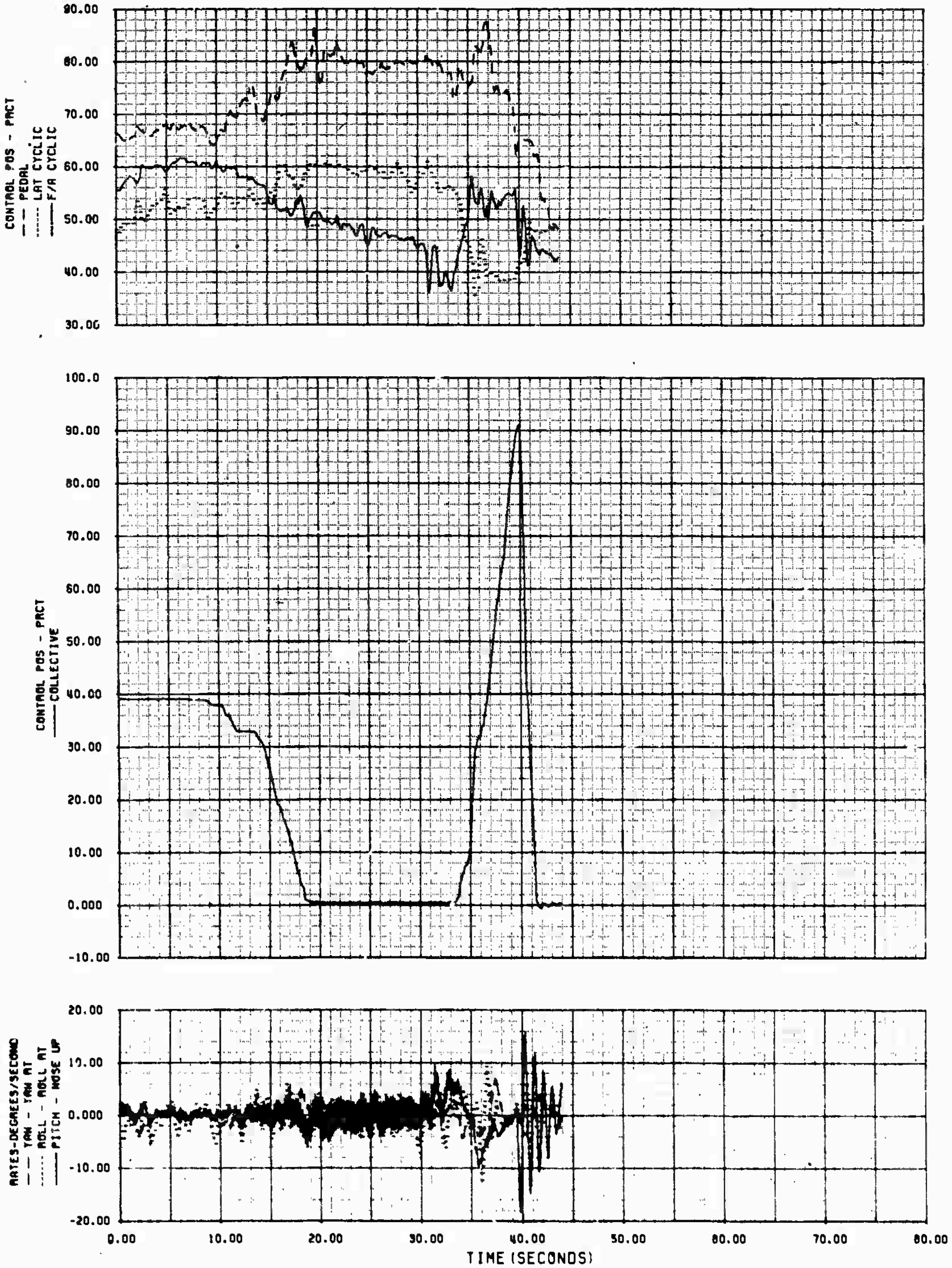


FIGURE 58C
AUTOROTATIONAL APPROACH AND LANDING
OH-58D USA S/N 69-16285

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
4328	110.9	1609	22.0	396	55	AUTOROTATION

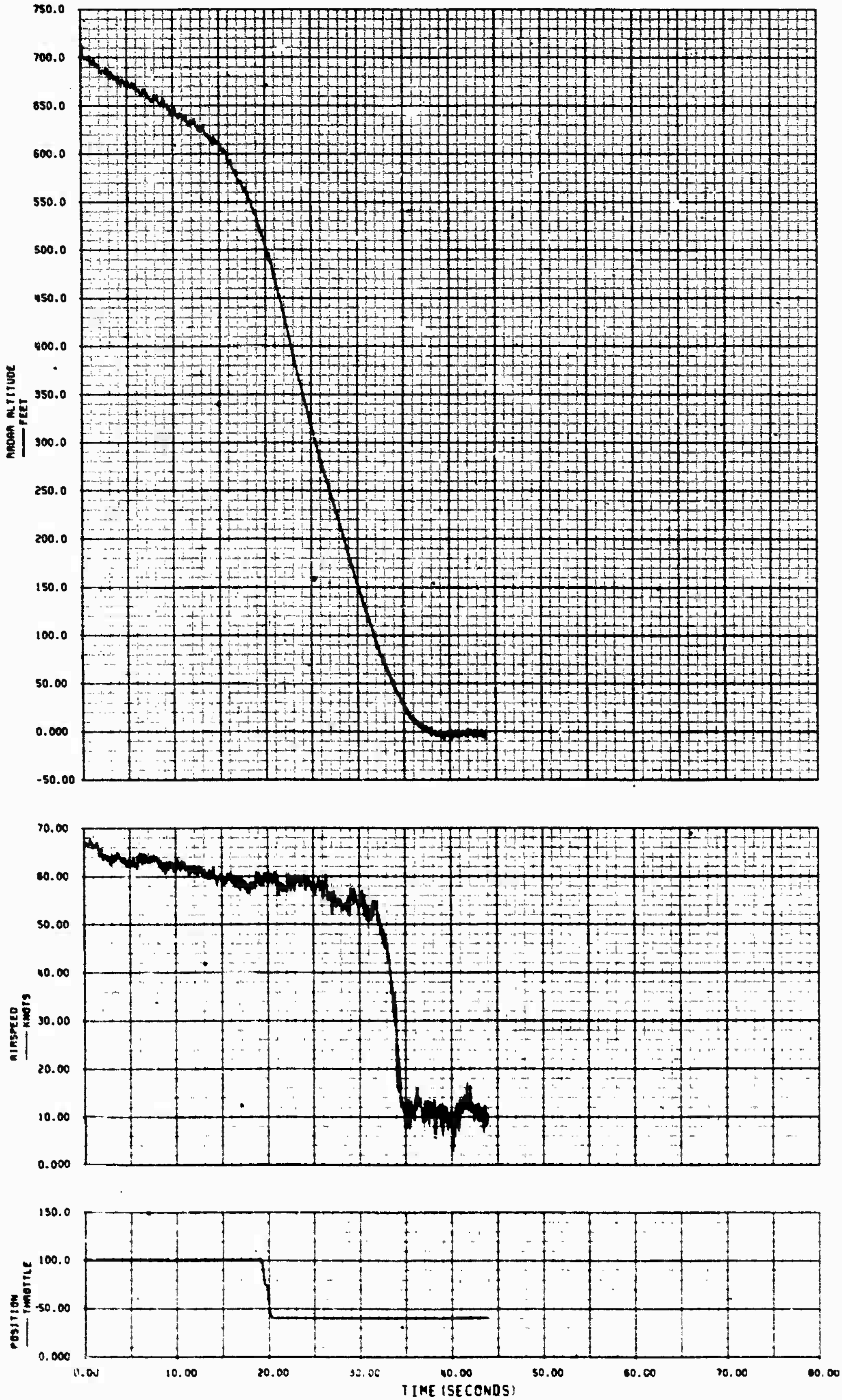


FIGURE 59

VIBRATION CHARACTERISTICS
OH-58D USA S/N 69-16285
PILOT SEAT

AVG GROSS WEIGHT	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)
3850	112.5 (AFT)	5470	17.0	395

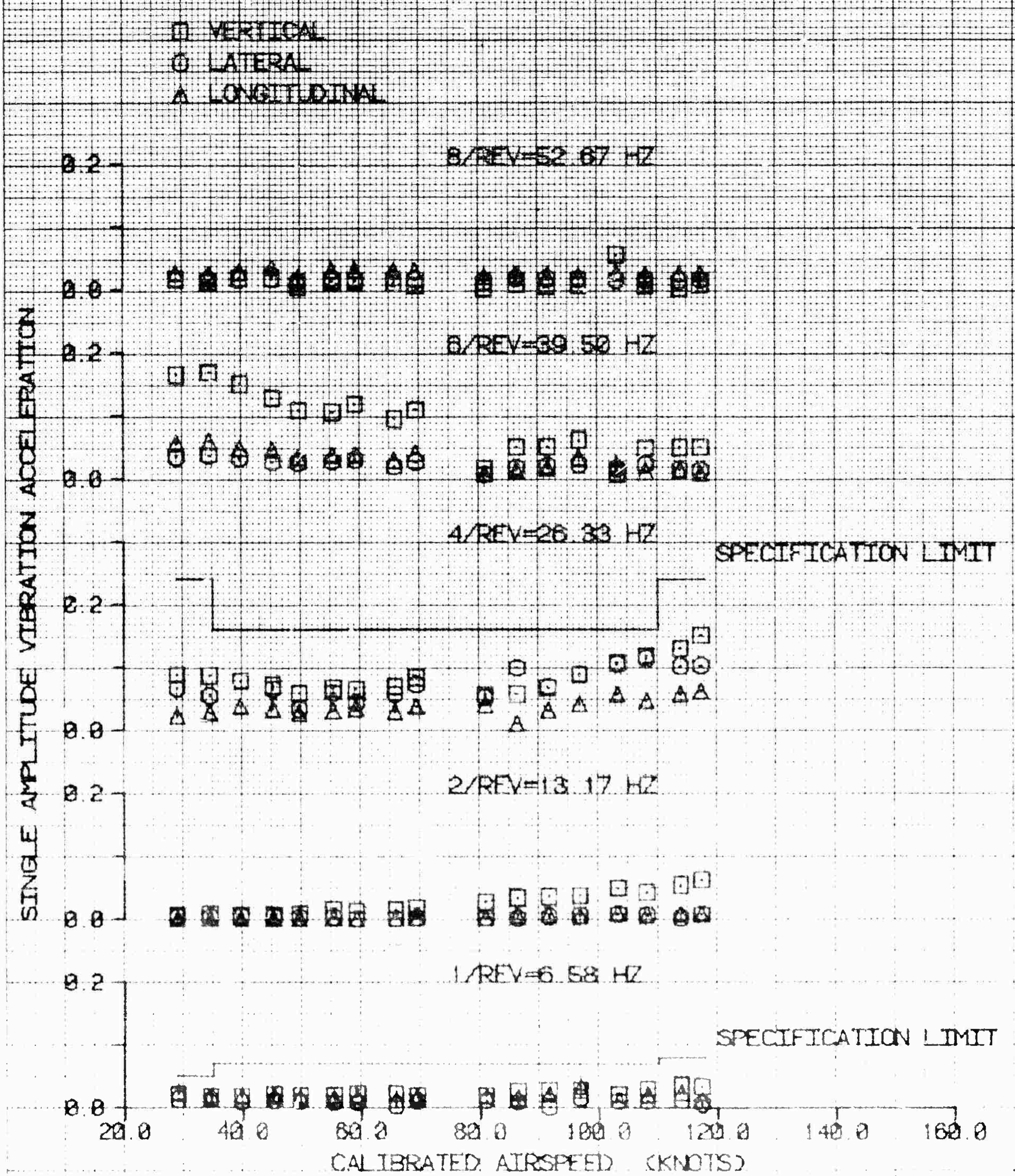


FIGURE 60

VIBRATION CHARACTERISTICS
OH-58D USA S/N 69-16285
CO-PILOT SEAT

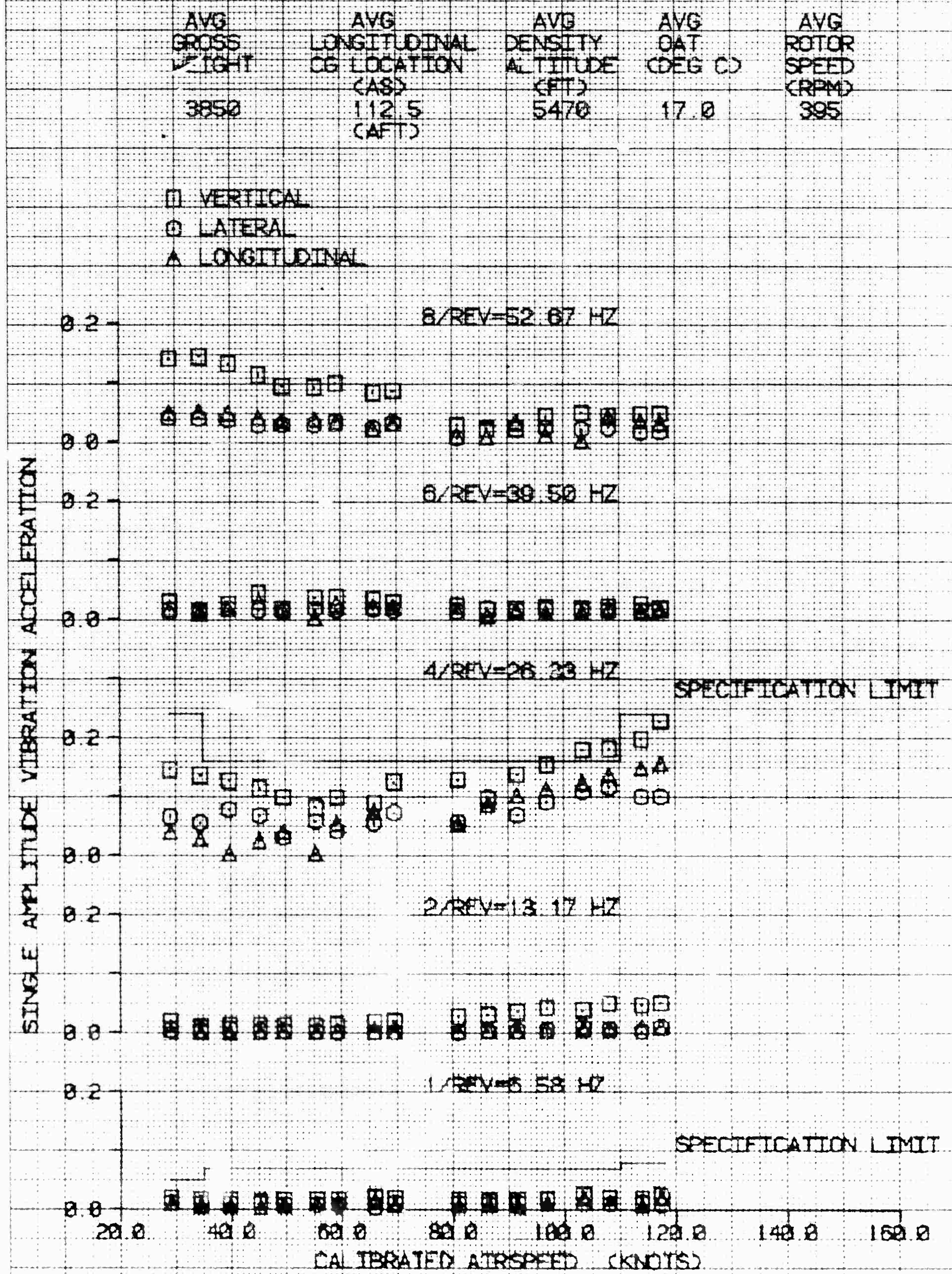


FIGURE 61

VIBRATION CHARACTERISTICS
OH-58D USA S/N 69-16285
CENTER OF GRAVITY

AVG GROSS WEIGHT	AVG LONGITUDINAL CG LOCATION (AS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)
3850	112.5 (AFT)	5470	17.0	395

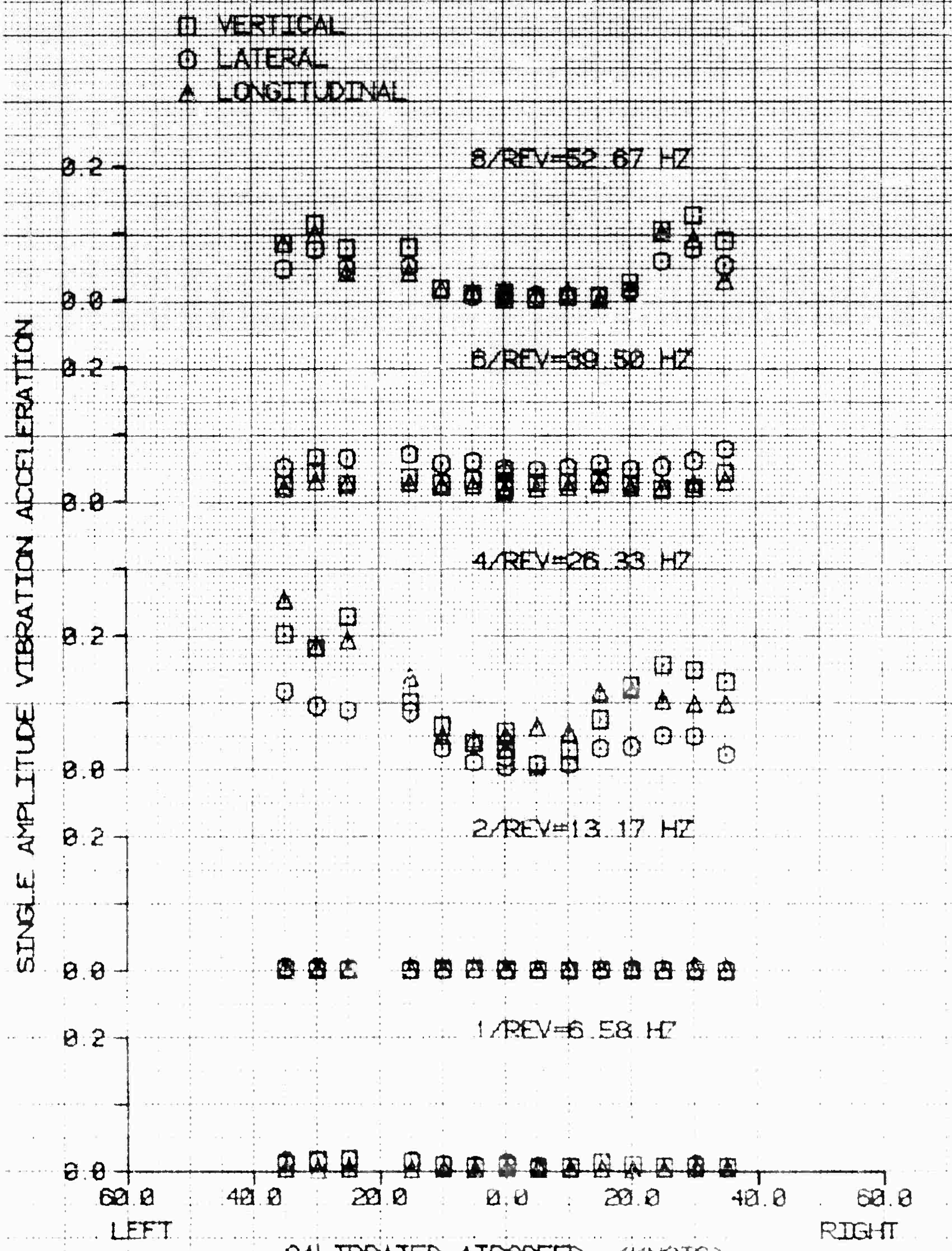
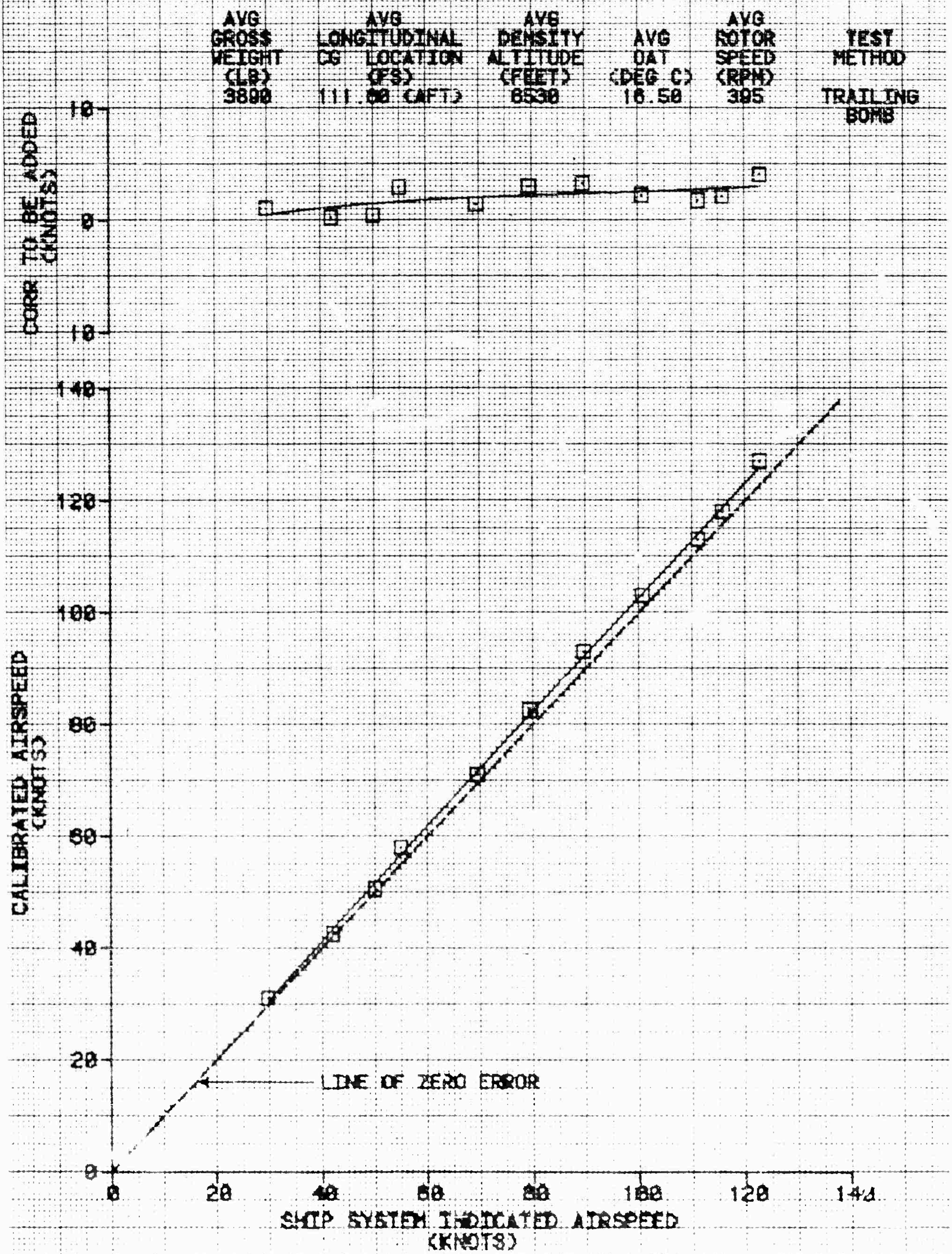


FIGURE 62
SHIP SYSTEM AIRSPEED CALIBRATION
OH-58D USA S/N 69-16285
LEVEL FLIGHT



APPENDIX F. PHOTOS

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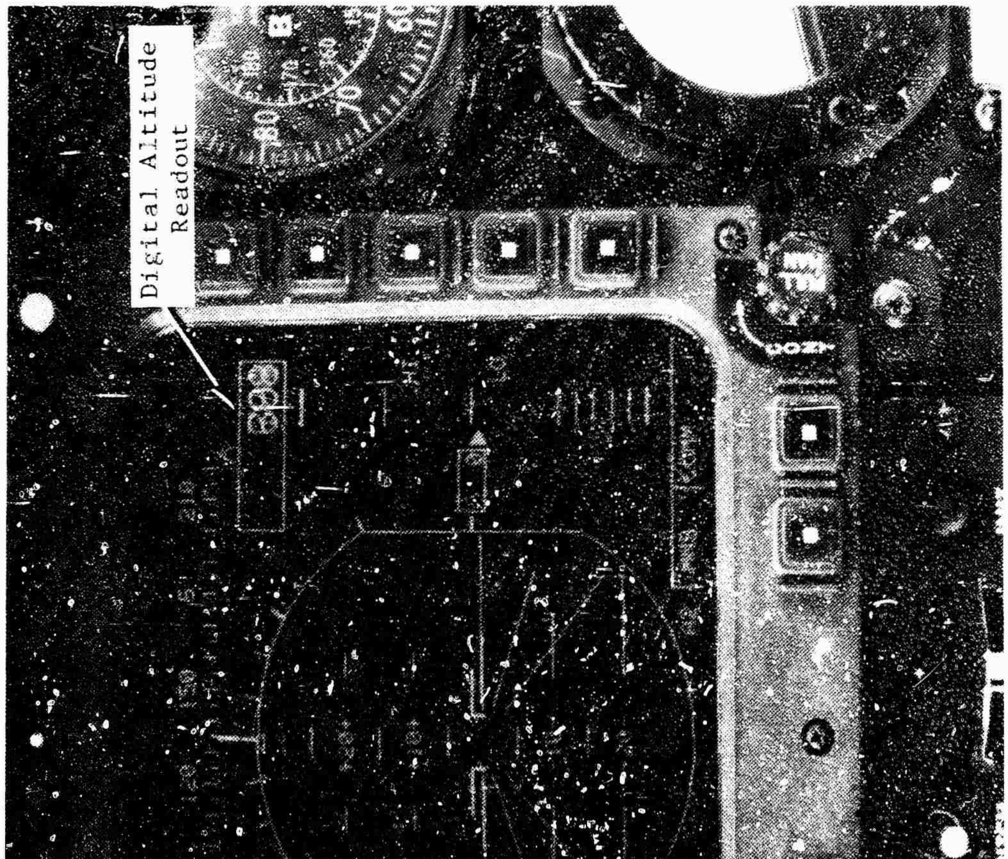

Photo 1. Multifunction Display

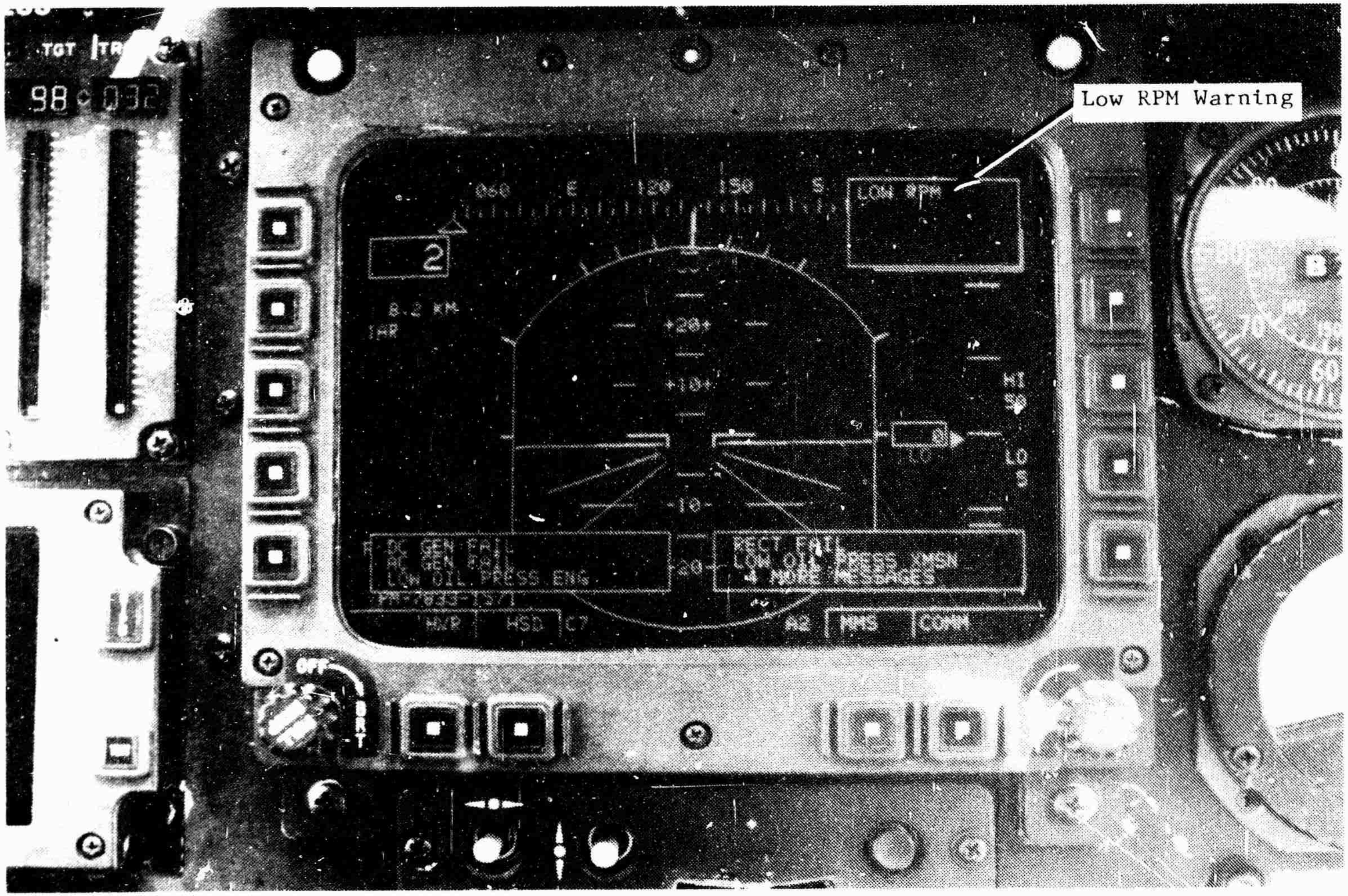

Photo 2. Multifunction Display with Warning Indication

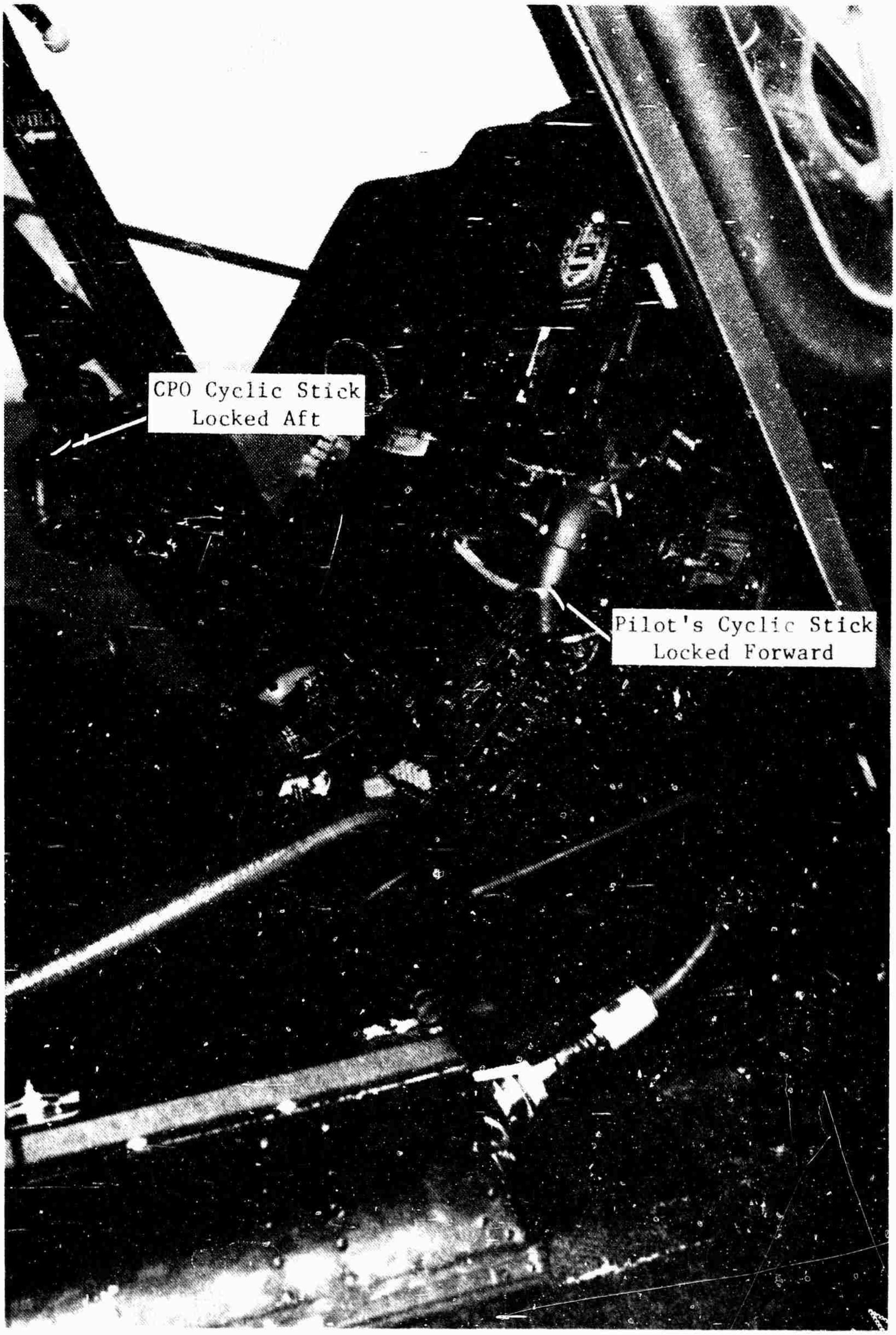

Photo 3. Misalignment of Pilot and CPO Cyclic Sticks

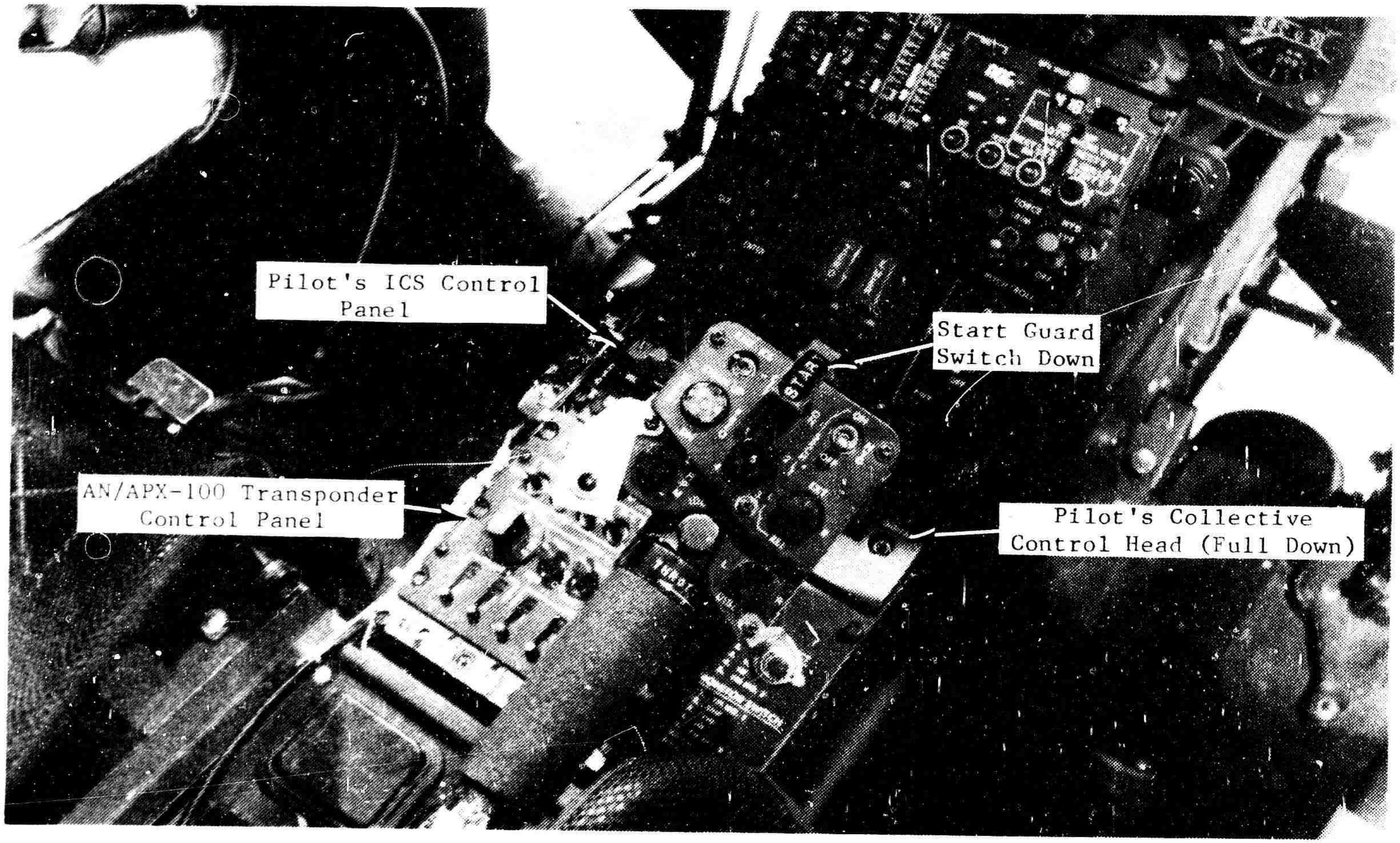

Photo 4. Blocked View of Transponder and Pilot's ICS Control Panel with Collective Full Down

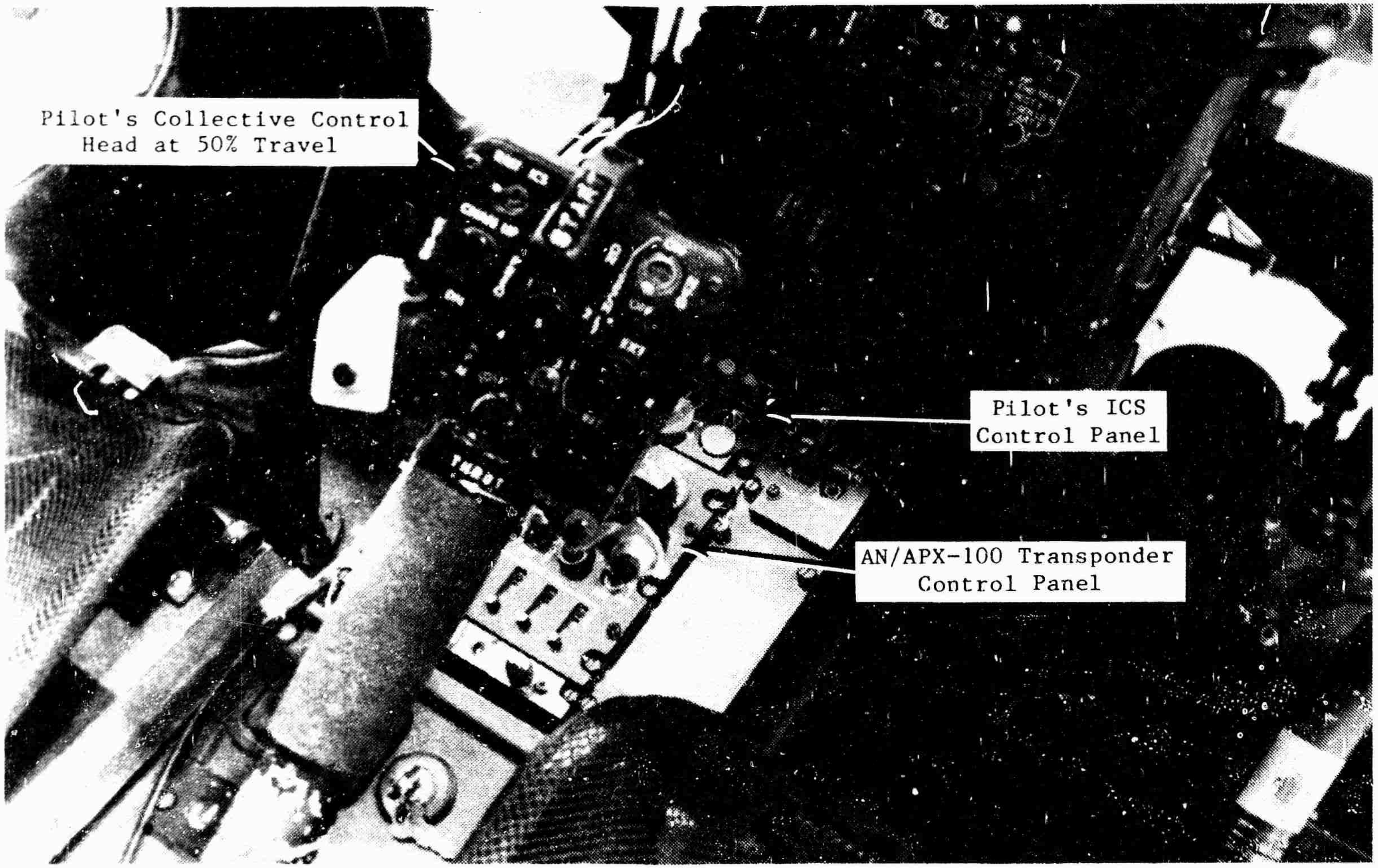

Photo 5. Blocked View of Transponder and Pilot's ICS Control Panels with Collective at 50% Travel

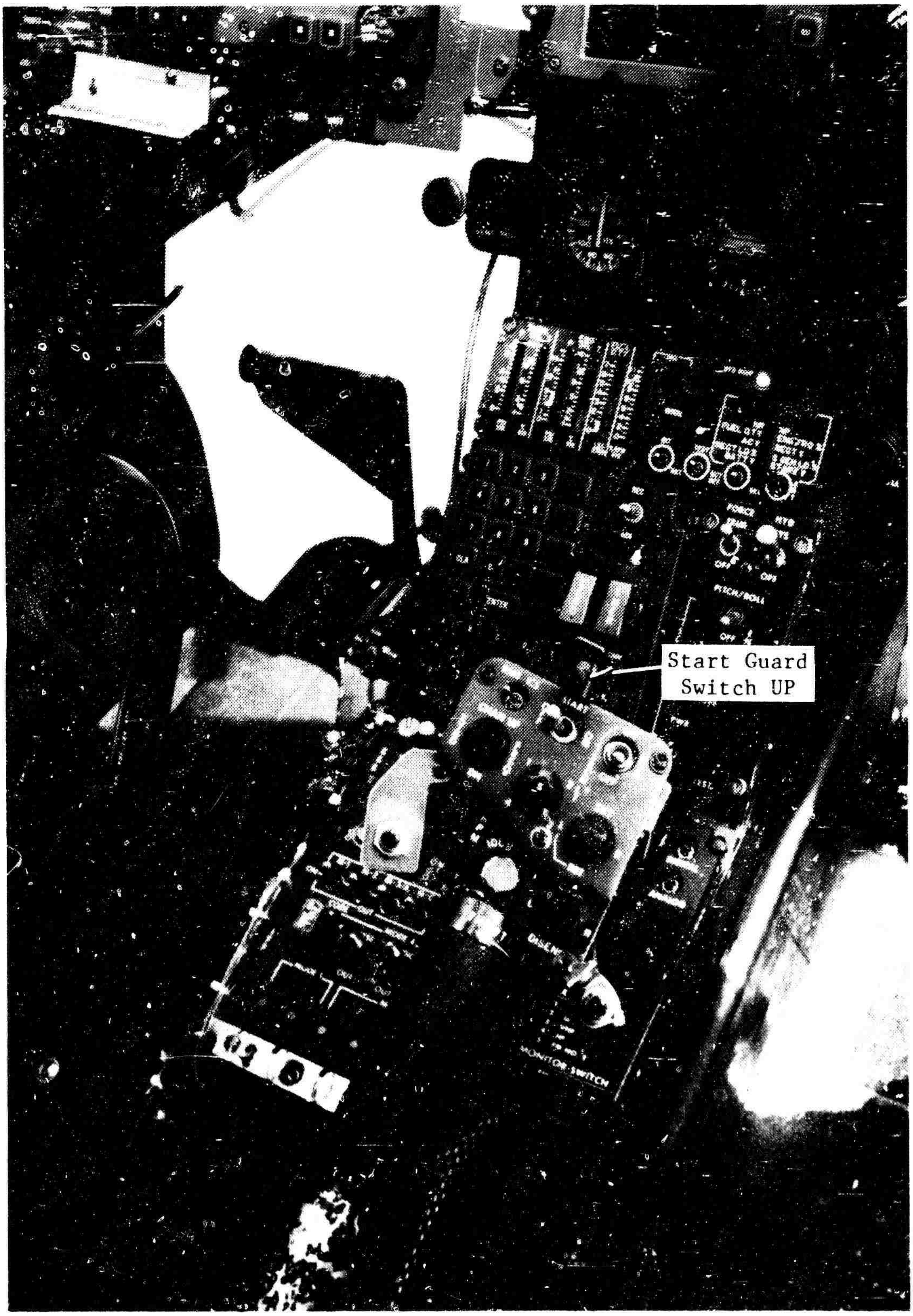

Photo 6. Pilot's Collective Control Head with the Start Guard Switch UP

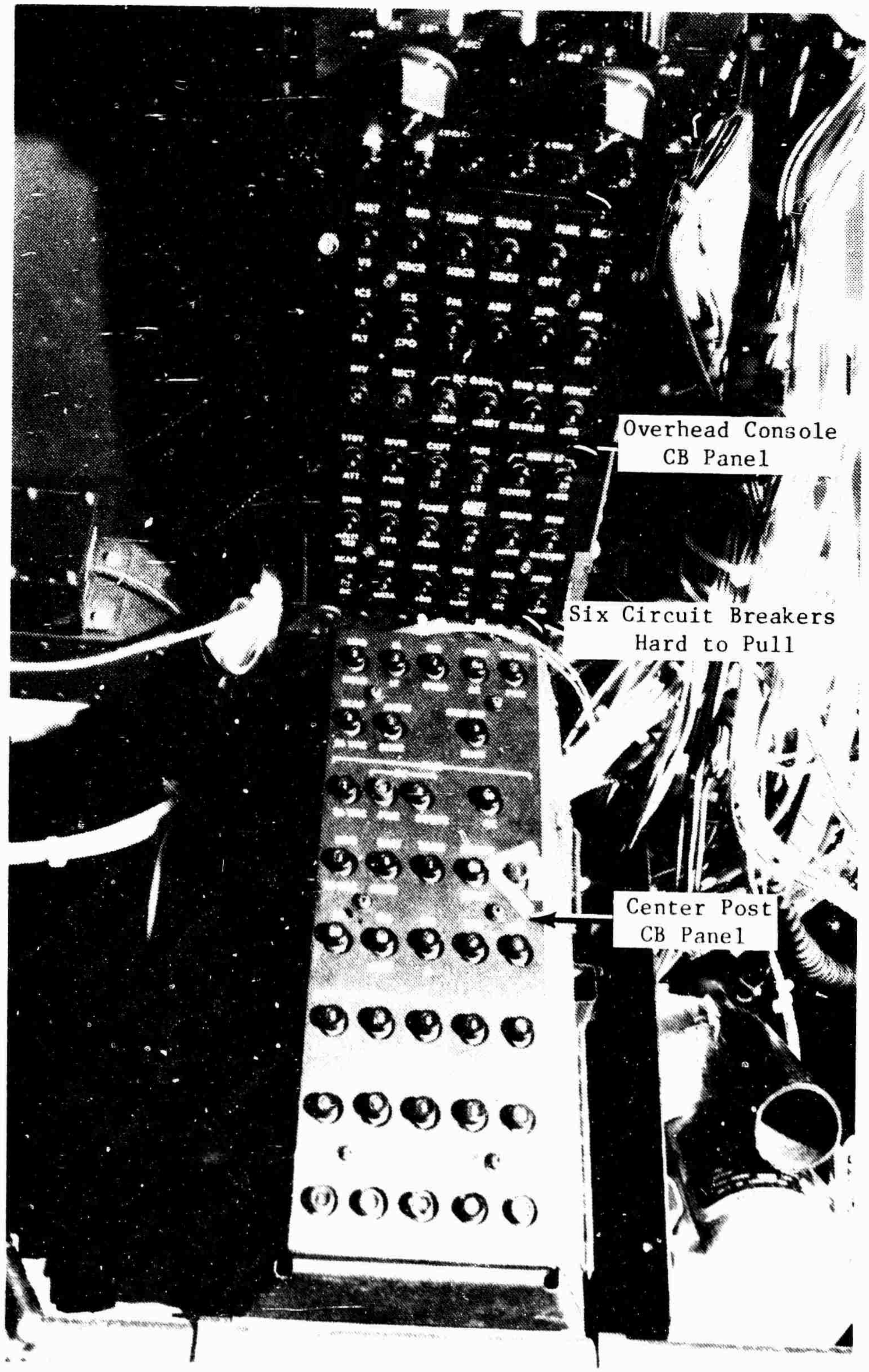

Photo 7. Circuit Breaker Panels

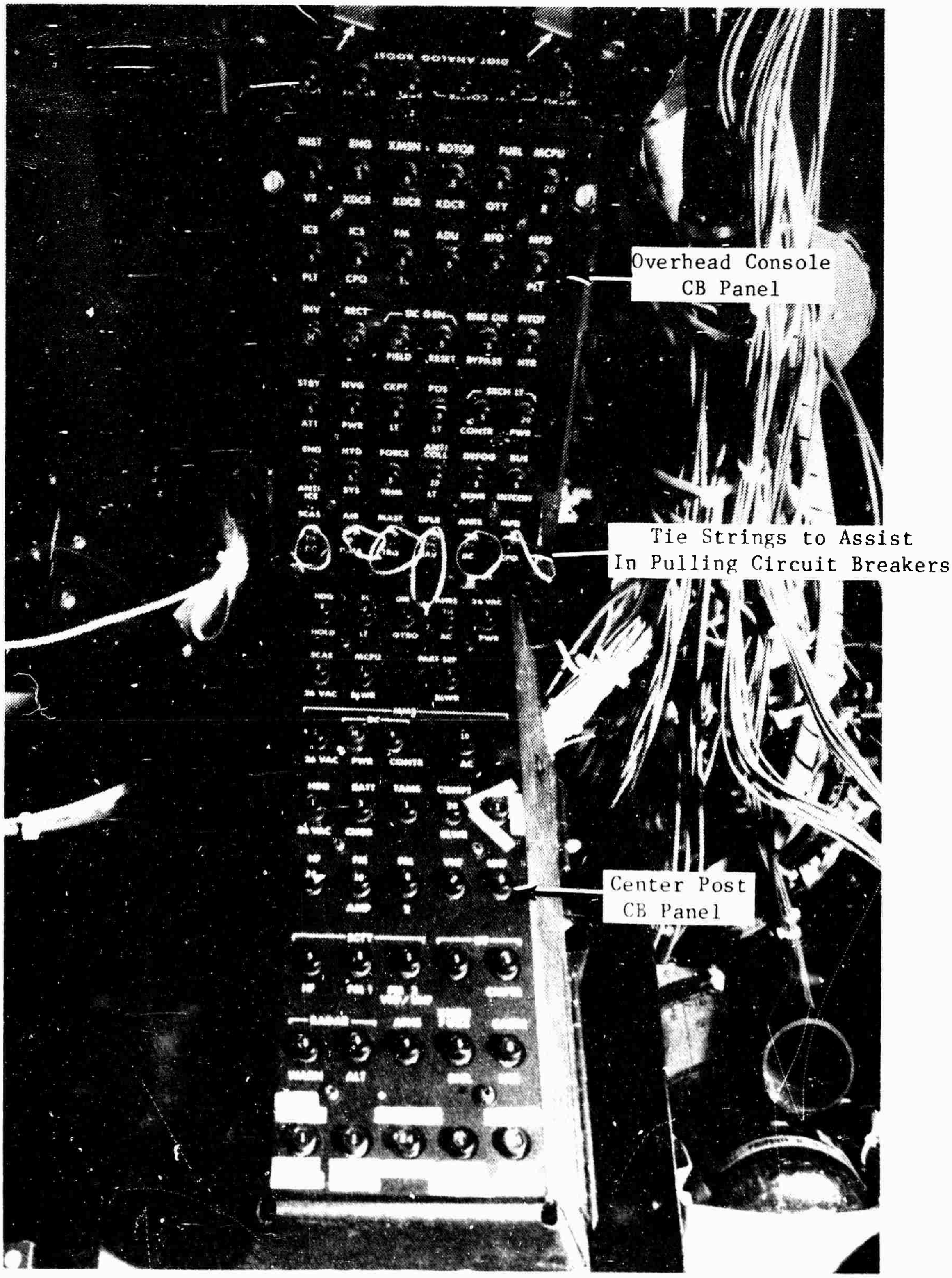

Photo 8. Circuit Breaker Panels

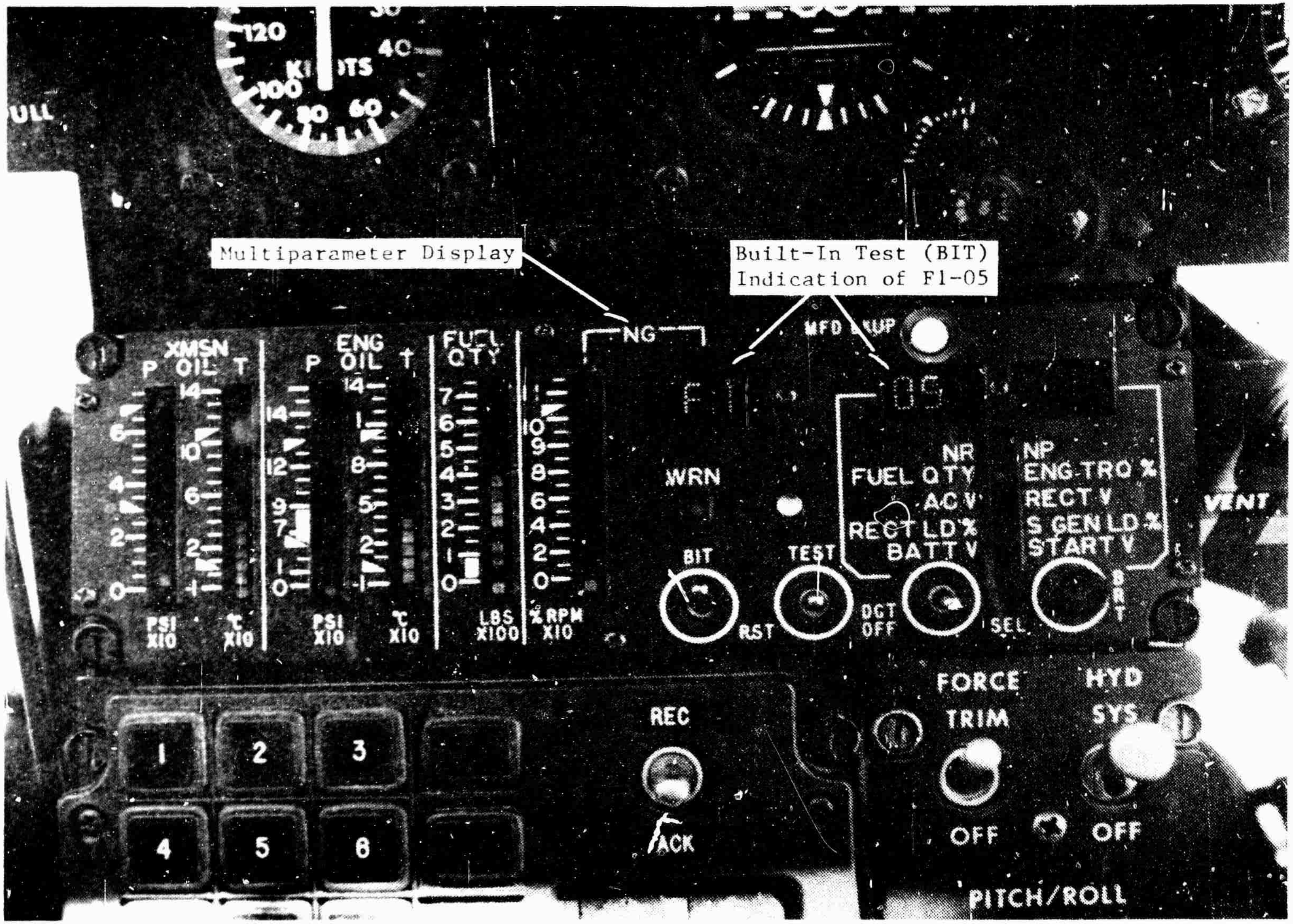

Photo 9. Multiparameter Display with Built-In Test Error Indication

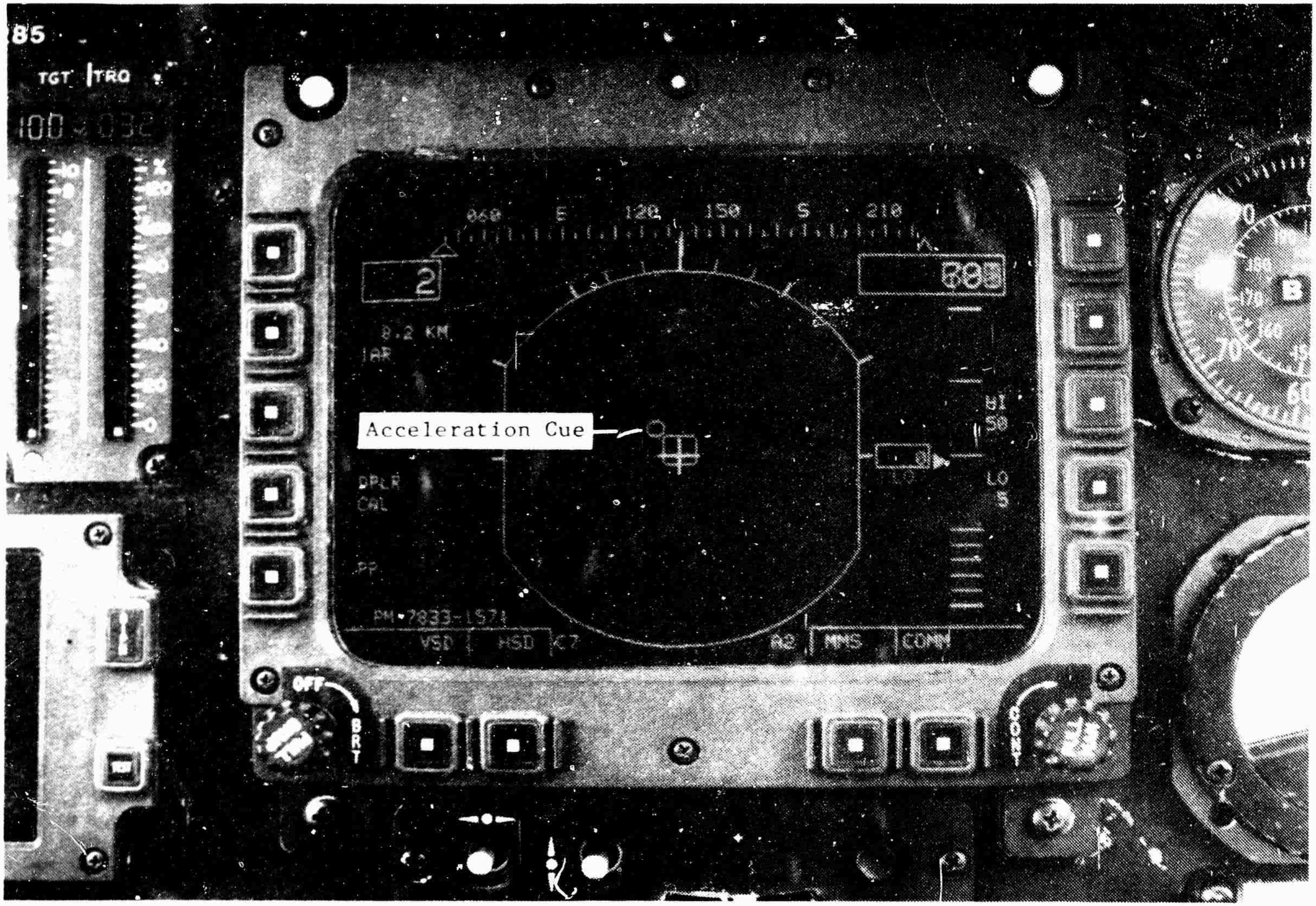

Photo 10. Multifunction Display (MFD) in Hover/Bob-up Mode

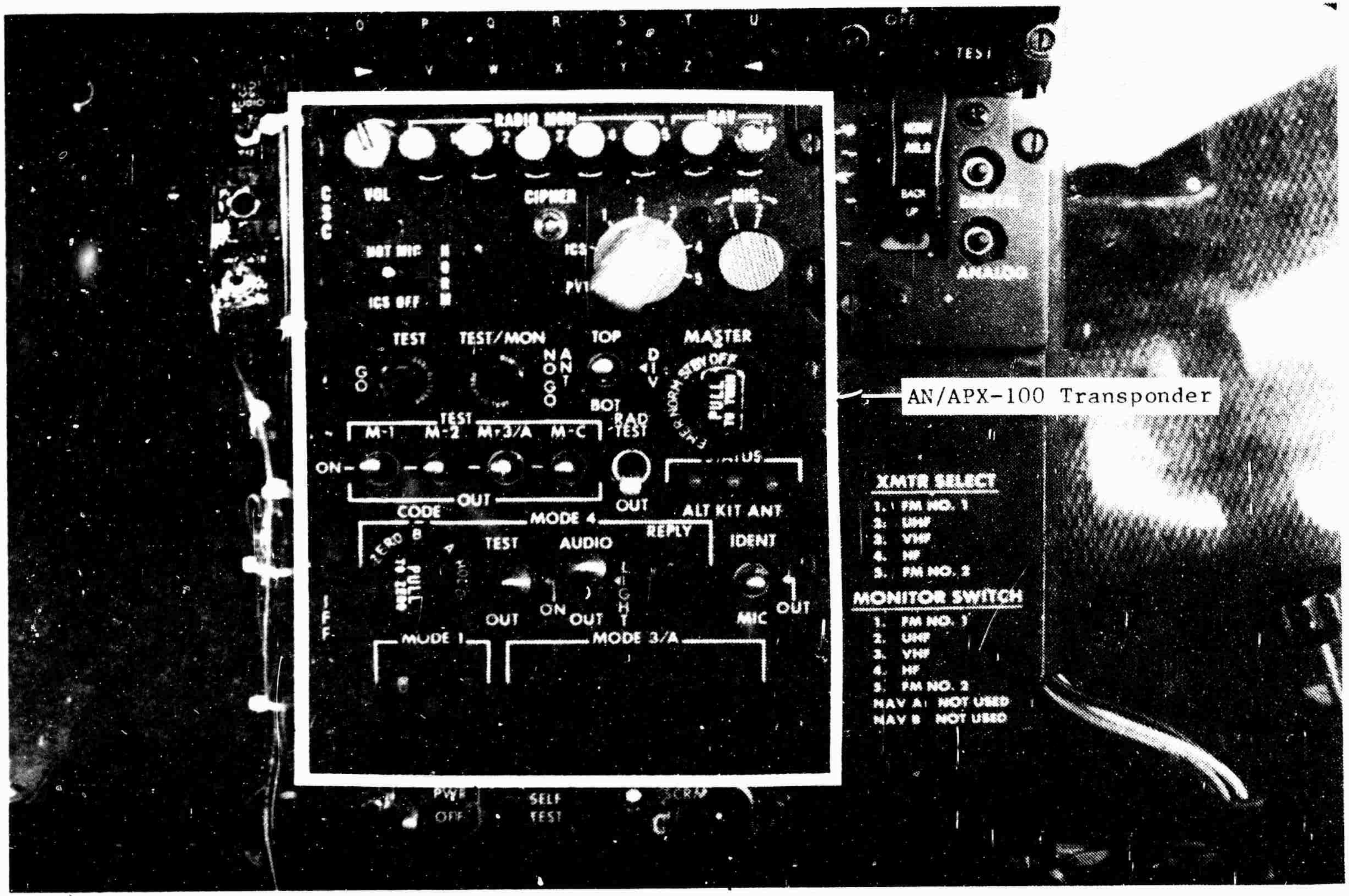

Photo 11. AN/APX-100 Transponder

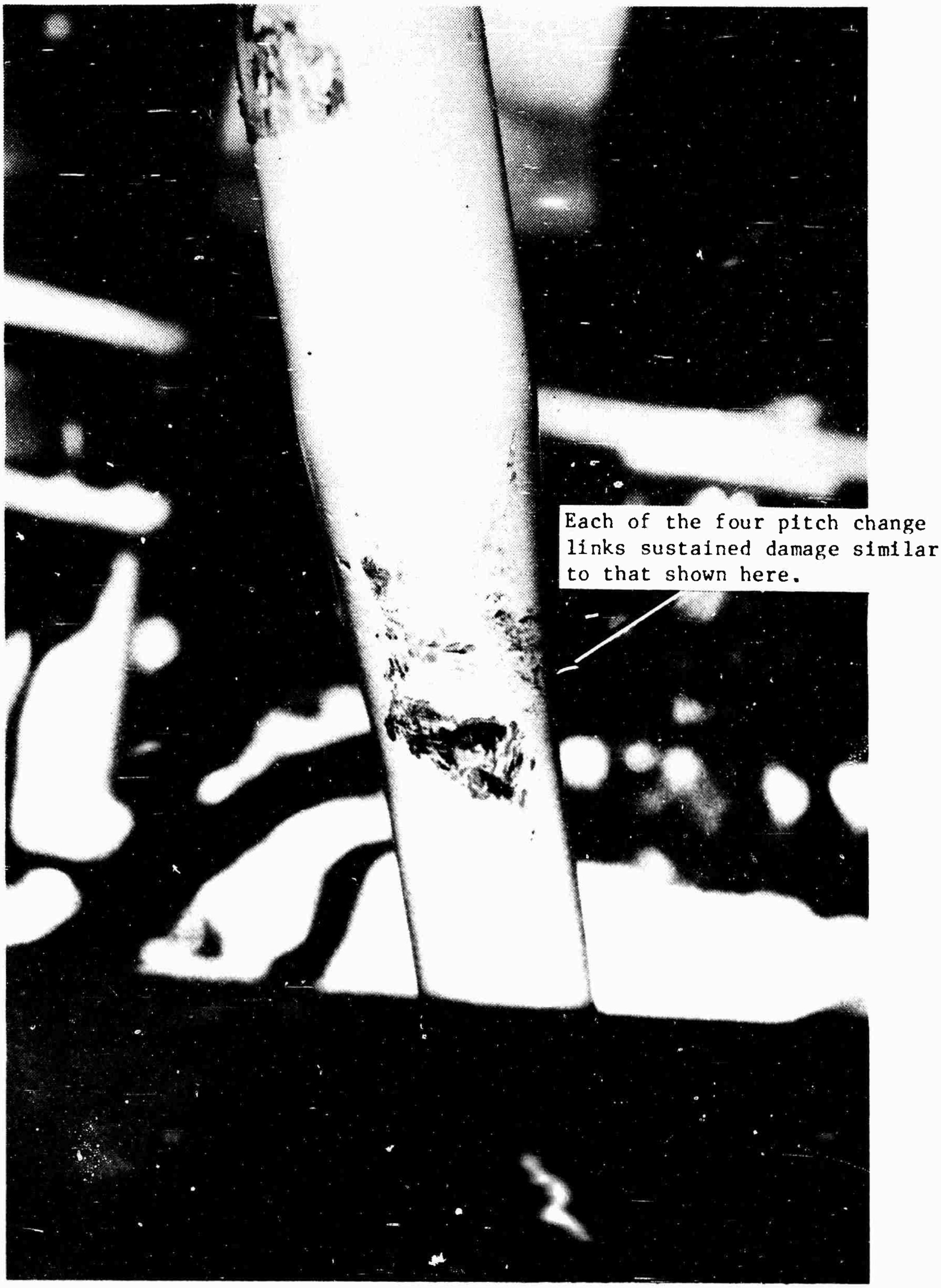

Photo 12. Pitch Change Link Damage

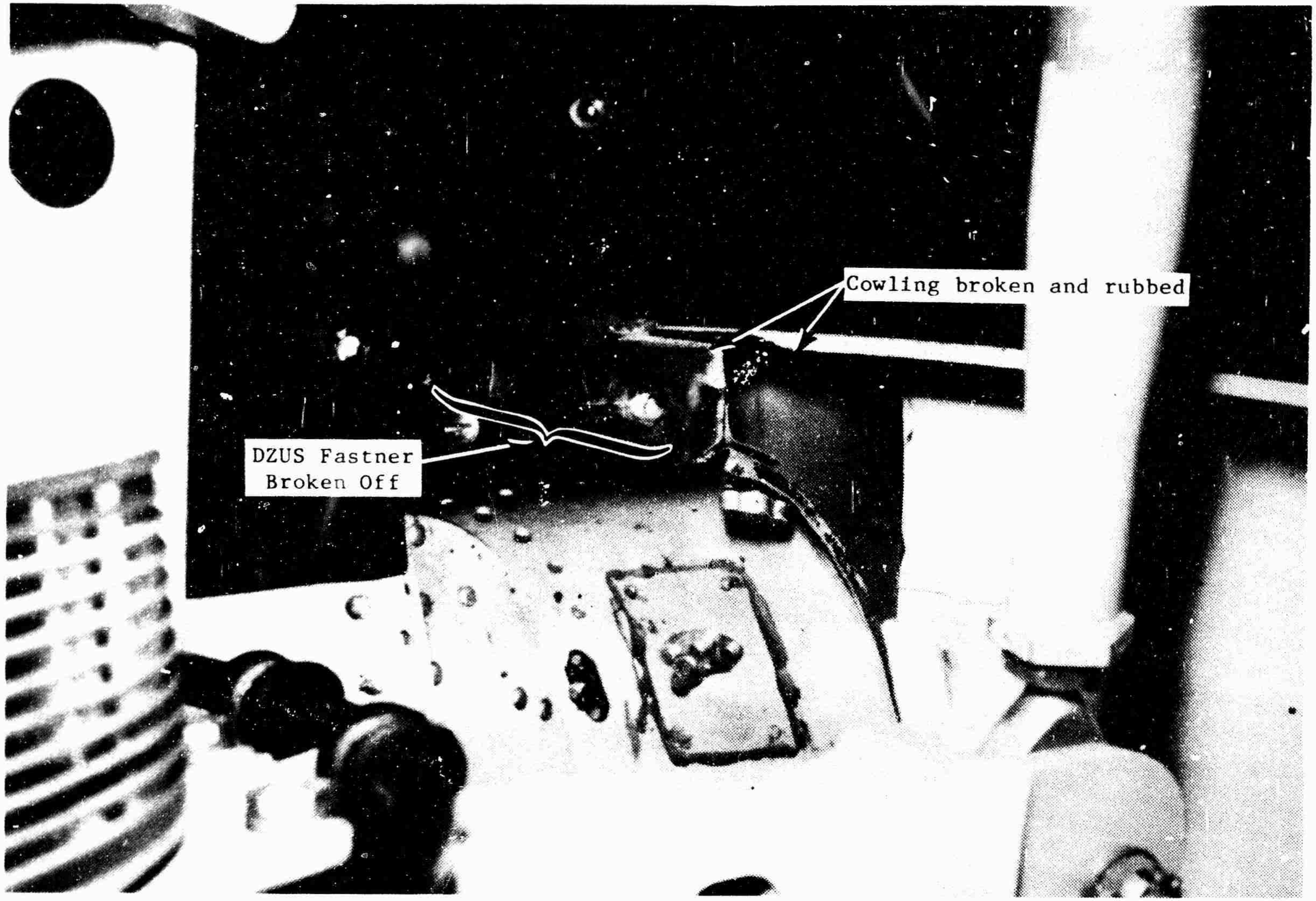

Photo 13. Cowling Dama;e

APPENDIX F. EQUIPMENT PERFORMANCE REPORTS

The following EPR's were submitted:

Number	Subject
83-26-1	Erroneous Activation of High RPM Warning*
83-26-2	Erroneous Activation of Mast Overtorque Warning*
83-26-3	Pitch/Roll SCAS Disengagements
83-26-4	Erroneous Activation of Engine Out Warning*
83-26-5	RFD Malfunction and Hard to Read
83-26-6	MCPU Resets
83-26-7	Oil Seep in T/R Gearbox*
83-26-8	Engine Start Problems
83-26-9	Erroneous Activation of High RPM Warning During Ground Run-up
83-26-10	De-synchronization of Radio Frequencies
83-26-11	MCPU Resets
83-26-12	Main Rotor Pitch Link/Cowling Contact*

*Problem was corrected during the evaluation.

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